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SELECTED DATA FROM A TRANSONIC FLEXIBLE  
WALLED TEST SECTION

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SELECTED DATA FROM A  
TRANSONIC FLEXIBLE WALLED TEST SECTION

by

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This is a semi-annual Progress Report for the period to May, 1980, on work undertaken on NASA Grant NSG-7172 entitled "The Self Streamlining of the Test Section of a Transonic Wind Tunnel". The Principal Investigator is Dr. M.J. Goodyer.

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## 1. TRANSONIC SELF-STREAMLINING WIND TUNNEL DATA

During the course of over two hundred and fifty test runs of the Transonic Self-Streamlining Wind Tunnel (TSWT) at Southampton University UK<sup>2</sup>, twenty-four runs were performed with the flexible walls 'streamlined' around a two-dimensional NACA 0012-64 section of 4 inch (10.16cm) chord, over the Mach number range 0.3 to 0.89. The purpose of this report is to present relevant wall and model data for the streamlined cases.

The practical interpretation of wall streamlining requires some explanation. Whilst the flexible walls can only be positioned within some tolerance set by experimental and theoretical considerations, good streamlining is achieved by reducing measures of wall streamlining quality below acceptable limits<sup>1</sup>. For TSWT, these measures are:-

- i)  $E$  for each flexible wall which is the average of the modulus of the imbalance in pressure coefficient between real and imaginary flows.
- ii) Residual interferences due to the flexible wall loading, in terms of induced angle of attack, induced camber and a streamwise velocity error at the model.

Experience has shown that for good streamlining  $E$  should be less than 0.01 and that none of the three components of the residual interferences should induce an error in the model  $C_L$  greater than 0.008.

From Table 1, it can be seen that for runs above approximately Mach 0.85, the flexible walls may not be good streamlines. However, they are the best the current wall setting strategy will allow, based on linearised compressible theory. Although  $E$  was greater than 0.01 for these runs, the residual interferences were still acceptably small.

The summary of streamlined wall data in Table 1 gives further information on the operating experience with TSWT and the summarised runs demonstrate the repeatability of results obtained using different streamlining 'paths' for Mach numbers up to 0.7 (as shown by data from Runs 72 and 63). Averages of four iterations from straight walls to streamlined walls and two iterations from contoured walls to streamlined

walls have been demonstrated. Either Mach number, angle of attack, transition strip or combinations thereof were changed from one streamlining cycle to the next. These changes were, in magnitude, typical of those which would normally be made during aerodynamic tests.

For each run with the model installed listed in Table 1, the airfoil pressure distribution is tabulated in Table 2 and plotted in Figure 1 in the order shown in Table 1. The accompanying airfoil force and pitching moment coefficients are calculated from the integrated airfoil pressure distributions. In addition, all the Mach number distributions along the centreline of each flexible wall are plotted in Figure 2, together with wall contours in Figure 3, except for those runs marked with an asterisk. The wall contours shown are the effective aerodynamic contours, also called delta \* contours. These are wall movements, positive up, corrected for changes in wall boundary layer displacement thickness between the run and the empty test section constant Mach number (straight wall) contours derived experimentally for a similar free stream Mach number.

The geometric angle of attack  $\alpha$  may not be related closely to aerodynamic angle of attack, and therefore changes in  $\alpha$  may be more aerodynamically meaningful than absolute values. A summary of the normal lift curve slopes for the streamlined wall data is shown as Figure 16 in Reference 1. The TSWT data is not corrected for residual interferences. Similarly, the NASA reference data has not been corrected using conventional correction techniques for a ventilated test section, but the model to test section height ratio was 4.75.

The Mach number distributions show reasonably well the extent of the regions of supercritical flow on each flexible wall at high transonic speeds, but there is evidently a need for more wall pressure tappings in the vicinity of the model. While the reflection of model shocks from the walls does not seem to be a problem, spark schlieren pictures<sup>1</sup> have shown the existence of significant shock wave/wall boundary layer interactions.

Preliminary investigations have been made at Mach 0.89 of modifications to the flexible wall shape around the shock on the top wall, to take some account of the boundary layer thickening on the wall. A localised hollow was introduced into the top wall, the depth

of the hollow being determined by simple shock/boundary layer theory<sup>3</sup>(the maximum displacement was 0.03 inch (0.76 mm) ). This adjustment has produced favourable effects on the airfoil pressure distribution obtained earlier<sup>1</sup> in terms of a movement of the lower pressure surface shock forward by 5% chord and an increase in the pressure coefficients over the aft half of the upper suction surface by .05 to 0.1. See Run 224 data.

There has been some effort to simulate in TSWT a portion of the imaginary flowfield above the test section, to assist with the development of imaginary flowfield computations. With an empty test section, the bottom wall effective aerodynamic contour was adjusted to match that of the top wall effective aerodynamic contour obtained from an earlier TSWT run with the model installed. The top wall was streamlined normally for each bottom wall shape. It was found that some further adjustment of the bottom wall was necessary to generate the required velocity distribution along the wall, apparently to allow for shock/boundary layer interactions. Localised hollows in the vicinity of the model have been used; the step increase in  $\delta^*$  at the shock was not extended downstream. Runs 208, 215 and 219 generated the best velocity distributions along the bottom wall for freestream Mach numbers of 0.89, 0.84 and 0.7 respectively. Note how the shock on the bottom wall in the imaginary flowfield simulation at Mach 0.89 is in good position agreement with the shock in the real flowfield over the airfoil.

Four sets of effective aerodynamic straight wall contours have been derived experimentally using an empty test section for Mach numbers of 0.3, 0.5, 0.7 and 0.9. For these contours, allowances have been made for boundary layer growth on the walls, so that the velocity along the walls is near constant. For Mach numbers below 0.7, the walls were adjusted entirely in accordance with wall setting strategy demands. At Mach 0.9 final wall adjustment had to be intuitive, since the local wall Mach numbers were very sensitive to wall movement and the wall setting strategy was inadequate. The Mach number distributions along each wall are shown for the Mach 0.3 and 0.9 straight wall cases in Figure 2.29. This plot illustrates the difficulty

in setting 'straight walls' at high Mach numbers, although Run 195 data may not represent the best data possible. From Table 1, it can be seen that for both Run 30 and Run 195 curved flow was generated in the test section. It may prove necessary to eliminate this curvature to apply accurate model corrections for residual interferences.

Most of the data in this report has already been summarised and discussed<sup>1</sup>, but is presented here as a comprehensive library of numerical and graphical data which may prove useful to others engaged in the evaluation, design and use of transonic flexible walled test sections.

# LIST OF SYMBOLS

C	Model chord
CC	Chordwise force coefficient
CD	Pressure drag coefficient
CL	Life coefficient
CM	Pitching moment coefficient about the leading edge
CN	Normal force coefficient
$C_p$	Pressure coefficient
$C_p^*$	Sonic pressure coefficient
E	Average of the modulus of the pressure error between real and imaginary flows along a flexible wall.
$E_{AV}$	Average of top and bottom wall E
$E_{TOP}$	Top wall E
M	Freestream Mach number
x	Chordwise position
$\alpha$	Angle of attack (Geometric)
$\delta^*$	Boundary layer displacement thickness

REFERENCES.

1. M.J. Goodyer and S.W.D. Wolf 'The Development of a Self-Streamling Flexible Walled Transonic Test Section' AIAA Paper 80-0440, Mar 1980
2. S.W.D. Wolf and M.J. Goodyer 'Self Streamlining Wind Tunnel - Low Speed Testing and Transonic Test Section Design' NASA CR-145257, Oct 1977
3. B.I.F. Mason 'Development of a program for the flexible wall tunnel at transonic speeds' B.Sc. Honours project, University of Southampton, May 1980.

STREAMLINED WALLS

Figure No.	Run No.	Model $\alpha$	Mach No.	Iterations from straight walls	Iterations from contoured walls	Changes from contoured walls		EAV	Max. residual error $\Delta C_L$	Grit on
						$\Delta\alpha$	$\Delta M$			
1	184	4.0	0.890	-	Three	0	0.02	.0138	.0034	Yes
2	176	2.0	0.891	-	Two	0	0.025	.0190	.0021	Yes
3	108(M)	0	0.866	-	Two	0	0.112	.0123	.0031	No
4	168	4.5	0.846	-	Two	0.5	0	.0057	.0024	No
5	170	4.5	0.849	-	Two	0	0	.0068	.0035	Yes
6	172	2.0	0.848	-	Two	2.0	0	.0061	.0027	Yes
7	162	2.0	0.839	-	Two	2.0	0	.0067	.0043	No
8	100	2.0	0.84	-	Two	0	0.05	.008	.0072	No
9	136	0	0.84	-	One	-2.0	0	.0082	.0032	Yes
10	119/96	2.0	0.81	-	Two	0	0.1	.0063	.0047	No
11	188(S)*	0	0.796	-	Two	0	-0.05	.0078	.0043	Yes
12	105(M)	0	0.753	Three	-	-	-	.0072	.0032	No
13	*72	4.0	0.706	Four	-	-	-	.0062	.0013	No
14	*63	4.0	0.702	-	Two	1.0	0	.0035	.0037	No
15	*69	3.0	0.701	Four	-	-	-	.0045	.0026	No
16	*65	2.0	0.703	-	One	-2.0	0	.0043	.0049	No
17	93	2.0	0.712	-	One	0	0.2	.0075	.0032	No
18	122	0	0.698	-	Three	-2.0	-0.1	.0088	.008	No
19	115	6.0	0.506	-	Two	2.0	0	.0069	.0061	No
20	112	4.0	0.507	-	Two	2.0	0	.0045	.0051	No
21	91	2.0	0.508	-	One	0	0.2	.0045	.0009	No
22	109	2.0	0.504	-	Three	2.0	0	.005	.0046	No
23	105	0	0.506	Four	-	-	-	.0077	.0072	No
24	89	2.0	0.306	Three	-	-	-	.006	.0047	No
Special Cases				Remarks						
25	*224	4.0	0.882	Rerun of Run 184 with local hollow in top wall.				.0266	.0293	Yes
26	*208	-	0.889	Empty test section upper imaginary flowfield simulation for Run 184.				ETOP .0065	-	-
27	*215	-	0.841	Flowfield sim. for Run 162				EAV .0058	-	-
28	*219	-	0.708	Flowfield sim. for Run 72				.0052	-	-
29	*195	-	0.899	Empty test section				.0016 .0052	.0037	-
30	* 30	-	0.303	Empty test section				.0042 .0038	.0042	-
$\Delta C_L$ due to camber										

\* No plot of wall  $\delta^*$  contours available.

TABLE 2

NACA 0012-64 SECTION  
PRESSURE DISTRIBUTIONS AND FORCES

TABLE 2.1

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 184

ALPHA = 4.0

MACH NO. = 0.886

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 42

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2843	0.2843
1	-0.0849	0.6535
2	-0.5189	0.3663
5	-0.6969	0.1986
7	-0.7059	0.0819
9	-0.7045	0.0107
15	-0.6911	-0.0766
20	-0.7689	-0.1384
25	-0.7810	-0.2042
29	-0.8096	-0.2351
35	-0.8512	-0.2821
40	-0.8137	-0.3439
44	-0.5067	-0.4044
50	-0.3740	-0.4500
55	-0.3435	-0.4917
60	-0.3422	-0.5468
64	-0.3391	-0.5829
70	-0.3391	-0.6352
75	-0.3391	-0.6859
80	-0.3293	-0.7012
85	-0.3206	-0.7323
90	-0.3102	-0.7481
95	-0.2977	-0.5458

	UPPER	LOWER	TOTAL
CN	0.4983	-0.3658	0.1325
CC	-0.0073	0.0471	0.0398
CM	-0.2020	0.2537	0.0517

AIRFOIL PERFORMANCE

CL	CD	CM
0.1294	0.0489	0.0517

TABLE 2.2

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 176

ALPHA = 2.0

MACH NO. = 0.891

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 34

ZCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.3033	0.3033
1	0.1770	0.4296
2	-0.2166	0.1196
5	-0.3263	-0.0197
7	-0.3814	-0.1092
9	-0.4367	-0.1762
15	-0.4919	-0.2459
20	-0.5484	-0.2880
25	-0.5708	-0.3393
29	-0.5860	-0.3884
35	-0.6083	-0.4121
40	-0.6201	-0.4582
44	-0.6740	-0.5148
50	-0.7042	-0.5556
55	-0.7278	-0.5840
60	-0.7554	-0.6339
64	-0.7636	-0.6610
70	-0.4659	-0.7069
75	-0.3003	-0.7491
80	-0.2597	-0.6840
85	-0.2346	-0.2971
90	-0.2159	-0.2356
95	-0.1961	-0.2084

	UPPER	LOWER	TOTAL
CN	0.4702	-0.3988	0.0714
CC	0.0093	0.0293	0.0385
CM	-0.2175	0.2266	0.0091

AIRFOIL PERFORMANCE

CL	CD	CM
0.0700	0.0410	0.0091

TABLE 2.3

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 108

ALPHA = 0.0

MACH NO. = 0.866

WING DATA FILE NAME = \*WING3.DAT  
INPUT FILE NO. - 17

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2452	0.2452
1	0.3198	0.1707
2	0.0136	-0.1436
5	-0.1796	-0.2953
7	-0.2747	-0.3625
9	-0.3166	-0.4263
15	-0.3992	-0.5159
20	-0.4520	-0.5770
25	-0.4668	-0.6110
29	-0.4804	-0.6389
35	-0.5007	-0.6470
40	-0.5169	-0.6470
44	-0.5683	-0.7014
50	-0.6116	-0.7027
55	-0.6259	-0.6517
60	-0.5825	-0.6395
64	-0.5593	-0.6363
70	-0.5571	-0.5725
75	-0.5670	-0.3347
80	-0.3154	-0.1393
85	-0.0760	-0.0154
90	0.0561	0.0859
95	0.1316	0.1563

	UPPER	LOWER	TOTAL
CN	0.3722	-0.4220	-0.0498
CC	0.0099	0.0015	0.0114
CM	-0.1697	0.1730	0.0033

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0498	0.0114	0.0033

TABLE 2.4

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 168

ALPHA = 4.5

MACH NO. = 0.846

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 26

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2130	0.2130
1	-0.2467	0.6727
2	-0.5299	0.3888
5	-0.7401	0.1972
7	-0.7920	0.0700
9	-0.8214	-0.0042
15	-0.8325	-0.0966
20	-0.8940	-0.1596
25	-0.9149	-0.2254
29	-0.9314	-0.2618
35	-0.8853	-0.3136
40	-0.8447	-0.3668
44	-0.8699	-0.4242
50	-0.7818	-0.4691
55	-0.6167	-0.5023
60	-0.5063	-0.5455
64	-0.4378	-0.5574
70	-0.3777	-0.5228
75	-0.3153	-0.4967
80	-0.2540	-0.4843
85	-0.1871	-0.2314
90	-0.1253	-0.1049
95	-0.0652	-0.0754

	UPPER	LOWER	TOTAL
CN	0.5729	-0.2685	0.3044
CC	-0.0154	0.0308	0.0154
CM	-0.2163	0.1645	-0.0518

AIRFOIL PERFORMANCE		
CL	CD	CM
0.3023	0.0393	-0.0518

TABLE 2.5

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 170

ALPHA = 4.5

MACH NO. = 0.849

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 28

ZCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2905	0.2905
1	-0.0476	0.6286
2	-0.5217	0.3289
5	-0.8140	0.1617
7	-0.8214	0.0447
9	-0.8311	-0.0265
15	-0.7991	-0.1144
20	-0.8659	-0.1758
25	-0.8729	-0.2414
29	-0.9010	-0.2746
35	-0.9219	-0.3317
40	-0.4950	-0.3875
44	-0.3879	-0.4446
50	-0.3518	-0.4892
55	-0.3362	-0.5214
60	-0.3206	-0.5771
64	-0.3089	-0.6076
70	-0.2880	-0.6371
75	-0.2642	-0.6342
80	-0.2376	-0.3344
85	-0.2088	-0.2399
90	-0.1782	-0.2014
95	-0.1477	-0.1652

	UPPER	LOWER	TOTAL
CN	0.4752	-0.2992	0.1760
CC	-0.0152	0.0324	0.0172
CM	-0.1712	0.1824	0.0111

AIRFOIL PERFORMANCE

CL	CD	CM
0.1741	0.0309	0.0111

TABLE 2.6

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 172

ALPHA = 2.0

MACH NO. = 0.848

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 30

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2578	0.2578
1	0.1424	0.3732
2	-0.2847	0.0468
5	-0.4354	-0.0950
7	-0.4903	-0.1848
9	-0.5843	-0.2537
15	-0.5881	-0.3158
20	-0.6418	-0.3626
25	-0.6597	-0.4150
29	-0.6672	-0.4642
35	-0.6989	-0.4903
40	-0.7030	-0.5220
44	-0.7456	-0.5743
50	-0.7566	-0.6143
55	-0.6225	-0.6452
60	-0.4945	-0.6947
64	-0.2756	-0.7262
70	-0.1828	-0.5753
75	-0.1230	-0.2426
80	-0.0637	-0.1465
85	-0.0117	-0.0799
90	0.0347	-0.0173
95	0.0777	0.0397

	UPPER	LOWER	TOTAL
CN	0.4032	-0.3526	0.0506
CC	-0.0082	0.0134	0.0052
CM	-0.1445	0.1664	0.0219

AIRFOIL PERFORMANCE		
CL	CD	CM
0.0504	0.0070	0.0219

TABLE 2.7

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 162

ALPHA = 2.0

MACH NO. = 0.839

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 19

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2216	0.2216
1	0.0396	0.4037
2	-0.2787	0.0946
5	-0.4499	-0.0762
7	-0.5378	-0.1868
9	-0.5900	-0.2603
15	-0.6323	-0.3296
20	-0.6874	-0.3777
25	-0.7043	-0.4287
29	-0.7179	-0.4711
35	-0.7433	-0.4966
40	-0.7419	-0.5263
44	-0.7546	-0.5772
50	-0.7080	-0.6168
55	-0.6972	-0.6196
60	-0.7255	-0.5674
64	-0.5803	-0.5448
70	-0.2575	-0.5540
75	-0.1122	-0.1849
80	-0.0332	-0.0401
85	0.0326	-0.0080
90	0.1110	0.0601
95	0.1974	0.1448

	UPPER	LOWER	TOTAL
CN	0.4377	-0.3161	0.1216
CC	-0.0104	0.0092	-0.0012
CM	-0.1560	0.1384	-0.0176

AIRFOIL PERFORMANCE

CL	CD	CM
0.1216	0.0031	-0.0176

TABLE 2.8

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 100

ALPHA = 2.0

MACH NO. = 0.84

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. = 20

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2178	0.2178
1	0.0337	0.4018
2	-0.2838	0.1092
5	-0.4524	-0.0532
7	-0.5397	-0.1672
9	-0.5915	-0.2417
15	-0.6322	-0.3049
20	-0.6897	-0.3555
25	-0.7065	-0.4117
29	-0.7205	-0.4454
35	-0.7457	-0.4566
40	-0.7485	-0.4946
44	-0.7780	-0.5438
50	-0.7247	-0.6000
55	-0.7056	-0.6379
60	-0.7085	-0.6843
64	-0.6786	-0.6440
70	-0.3717	-0.4689
75	-0.1614	-0.2626
80	-0.0710	-0.1307
85	0.0034	-0.0443
90	0.0767	0.0529
95	0.1506	0.1449

	UPPER	LOWER	TOTAL
CN	0.4609	-0.3212	0.1397
CC	-0.0082	0.0118	0.0036
CM	-0.1724	0.1471	-0.0253

AIRFOIL PERFORMANCE		
CL	CD	CM
0.1395	0.0084	-0.0253

TABLE 2.9

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 136

ALPHA = 0.

MACH NO. = 0.84

WING DATA FILE NAME = \*WING3.DAT  
INPUT FILE NO. - 40

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2659	0.2659
1	0.3445	0.1873
2	0.0969	-0.0978
5	0.0125	-0.1446
7	-0.7449	-0.5901
9	-0.6813	-0.6494
15	-0.3497	-0.6053
20	-0.4160	-0.6577
25	-0.4229	-0.6949
29	-0.4545	-0.7418
35	-0.4944	-0.7625
40	-0.5165	-0.7460
44	-0.5702	-0.7515
50	-0.6074	-0.7956
55	-0.6322	-0.8246
60	-0.6418	-0.3778
64	-0.5905	-0.2401
70	-0.2731	-0.1690
75	-0.1881	-0.1080
80	-0.1248	-0.0537
85	-0.0649	-0.0151
90	-0.0011	0.0353
95	0.0576	0.0722

	UPPER	LOWER	TOTAL
CN	0.3504	-0.4138	-0.0634
CC	0.0051	-0.0054	-0.0003
CM	-0.1468	0.1502	0.0034

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0634	-0.0003	0.0034

TABLE 2.10

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 119

ALPHA = 2.0

MACH NO. = 0.81

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. - 39

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.1834	0.1834
1	-0.0417	0.4084
2	-0.3722	0.1054
5	-0.5390	-0.0639
7	-0.6269	-0.1742
9	-0.6745	-0.2412
15	-0.7028	-0.2963
20	-0.7519	-0.3395
25	-0.7519	-0.3842
29	-0.7569	-0.3936
35	-0.7718	-0.4218
40	-0.7183	-0.4500
44	-0.7019	-0.4678
50	-0.6677	-0.4515
55	-0.6683	-0.4208
60	-0.4931	-0.3911
64	-0.2602	-0.3584
70	-0.2223	-0.3469
75	-0.1783	-0.2680
80	-0.1114	-0.0976
85	-0.0379	-0.0385
90	0.0560	0.0446
95	0.1596	0.1433

	UPPER	LOWER	TOTAL
CN	0.4398	-0.2543	0.1856
CC	-0.0150	0.0078	-0.0072
CM	-0.1507	0.1097	-0.0410

AIRFOIL PERFORMANCE		
CL	CD	CM
0.1857	-0.0007	-0.0410

TABLE 2.11

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 188

ALPHA = 0.0

MACH NO. = 0.7957

WING DATA FILE NAME = \*WING4.DAT  
INPUT FILE NO. - 46

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.1367	0.1367
1	0.2794	-0.0060
2	-0.1554	-0.3404
5	-0.2181	-0.4724
7	-0.2973	-0.5124
9	-0.3302	-0.5722
15	-0.3603	-0.6245
20	-0.3705	-0.5498
25	-0.3765	-0.5916
29	-0.3874	-0.5654
35	-0.3949	-0.5444
40	-0.3904	-0.5159
44	-0.3979	-0.4904
50	-0.3830	-0.4334
55	-0.3554	-0.4020
60	-0.3210	-0.3583
64	-0.2804	-0.2847
70	-0.2321	-0.2260
75	-0.1771	-0.1626
80	-0.1112	-0.0876
85	-0.0423	-0.0151
90	0.0393	0.0671
95	0.1221	0.1390

	UPPER	LOWER	TOTAL
CN	0.2450	-0.3348	-0.0898
CC	-0.0007	-0.0127	-0.0134
CM	-0.0941	0.1120	0.0178

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0898	-0.0134	0.0178

TABLE 2.12

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 105

ALPHA = 0.0

MACH NO. = 0.7534

WING DATA FILE NAME = \*WING3.DAT  
INPUT FILE NO. - 14

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.1187	0.1187
1	0.2695	-0.0320
2	-0.0417	-0.3620
5	-0.2245	-0.4789
7	-0.2959	-0.5332
9	-0.3231	-0.5782
15	-0.3535	-0.5525
20	-0.3567	-0.5493
25	-0.3647	-0.5413
29	-0.3780	-0.5314
35	-0.3876	-0.5185
40	-0.3796	-0.5007
44	-0.3892	-0.4862
50	-0.3812	-0.4668
55	-0.3604	-0.4429
60	-0.3363	-0.4091
64	-0.3101	-0.3186
70	-0.2855	-0.2439
75	-0.2070	-0.1907
80	-0.1077	-0.1174
85	-0.0454	-0.0480
90	0.0370	0.0500
95	0.1330	0.1551

	UPPER	LOWER	TOTAL
CN	0.2457	-0.3398	-0.0941
CC	0.0005	-0.0118	-0.0112
CM	-0.0971	0.1183	0.0212

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0941	-0.0112	0.0212

TABLE 2.13

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 72

ALPHA = 4.0

MACH NO. = 0.7056

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. - 7

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.0222	-0.0222
1	-0.7746	0.7302
2	-1.1046	0.4540
5	-1.3217	0.2503
7	-1.3317	0.1146
9	-1.3163	0.0452
15	-1.1341	-0.0399
20	-1.1479	-0.0886
25	-0.5275	-0.1372
29	-0.5498	-0.1561
35	-0.5740	-0.1942
40	-0.5429	-0.2150
44	-0.5308	-0.2341
50	-0.4945	-0.2410
55	-0.4419	-0.2424
60	-0.3867	-0.2372
64	-0.3253	-0.2234
70	-0.2614	-0.2030
75	-0.1876	-0.1778
80	-0.1068	-0.1468
85	-0.0204	-0.0787
90	0.0710	-0.0042
95	0.1525	0.0660

	UPPER	LOWER	TOTAL
CN	0.4977	-0.0993	0.3983
CC	-0.0471	0.0192	-0.0280
CM	-0.1362	0.0639	-0.0723

AIRFOIL PERFORMANCE

CL	CD	CM
0.3993	-0.0001	-0.0723

TABLE 2.14

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 63

ALPHA = 4.0

MACH NO. = 0.702

WING DATA FILE NAME = \*WING1.DAT  
INPUT FILE NO. - 16

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.0321	-0.0321
1	-0.7952	0.7309
2	-1.1241	0.4573
5	-1.3433	0.2494
7	-1.3645	0.1166
9	-1.3334	0.0470
15	-1.1535	-0.0383
20	-1.1431	-0.0870
25	-0.5275	-0.1357
29	-0.5655	-0.1581
35	-0.5777	-0.1946
40	-0.5395	-0.2189
44	-0.5308	-0.2345
50	-0.4979	-0.2345
55	-0.4434	-0.2457
60	-0.3932	-0.2475
64	-0.3280	-0.2274
70	-0.2647	-0.2069
75	-0.1936	-0.1837
80	-0.1119	-0.1598
85	-0.0267	-0.1014
90	0.0641	-0.0120
95	0.1450	0.0598

	UPPER	LOWER	TOTAL
CN	0.5044	-0.1028	0.4016
CC	-0.0478	0.0196	-0.0282
CM	-0.1386	0.0667	-0.0719

AIRFOIL PERFORMANCE		
CL	CD	CM
0.4026	-0.0002	-0.0719

TABLE 2.15

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 69

ALPHA = 3.0

MACH NO. = 0.7012

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. - 4

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.0200	0.0200
1	-0.5428	0.5828
2	-0.9035	0.2897
5	-1.0162	0.0995
7	-1.0058	-0.0105
9	-0.9157	-0.0732
15	-0.7578	-0.1342
20	-0.6919	-0.1760
25	-0.6122	-0.2160
29	-0.5506	-0.2328
35	-0.5384	-0.2573
40	-0.5037	-0.2696
44	-0.4933	-0.2871
50	-0.4690	-0.2871
55	-0.4293	-0.2840
60	-0.3891	-0.2752
64	-0.3234	-0.2483
70	-0.2652	-0.2245
75	-0.1985	-0.1999
80	-0.1214	-0.1431
85	-0.0358	-0.0673
90	0.0589	-0.0007
95	0.1515	0.0751

	UPPER	LOWER	TOTAL
CN	0.4176	-0.1492	0.2684
CC	-0.0328	0.0136	-0.0192
CM	-0.1259	0.0767	-0.0492

## AIRFOIL PERFORMANCE

CL	CD	CM
0.2690	-0.0051	-0.0492

TABLE 2.16

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 65

ALPHA = 2.0

MACH NO. = 0.7028

WING DATA FILE NAME = \*WING1.DAT  
INPUT FILE NO. = 18

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.0549	0.0549
1	-0.2967	0.4065
2	-0.6264	0.1124
5	-0.7166	-0.0519
7	-0.7236	-0.1434
9	-0.6854	-0.1969
15	-0.6125	-0.2332
20	-0.5709	-0.2660
25	-0.5379	-0.2919
29	-0.5138	-0.3049
35	-0.4930	-0.3188
40	-0.4636	-0.3240
44	-0.4584	-0.3309
50	-0.4359	-0.3240
55	-0.3875	-0.3210
60	-0.3529	-0.3054
64	-0.3060	-0.2702
70	-0.2555	-0.2471
75	-0.1972	-0.2085
80	-0.1284	-0.1186
85	-0.0470	-0.0597
90	0.0477	0.0140
95	0.1544	0.0997

	UPPER	LOWER	TOTAL
CN	0.3552	-0.1956	0.1596
CC	-0.0215	0.0070	-0.0146
CM	-0.1146	0.0864	-0.0283

## AIRFOIL PERFORMANCE

CL	CD	CM
0.1600	-0.0090	-0.0283

TABLE 2.17

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 93

ALPHA = 2.0

MACH NO. = 0.7119

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. - 13

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.0673	0.0673
1	-0.2564	0.3910
2	-0.5856	0.0969
5	-0.6879	-0.0571
7	-0.7069	-0.1544
9	-0.6740	-0.2082
15	-0.6082	-0.2464
20	-0.5770	-0.2759
25	-0.5388	-0.3071
29	-0.5207	-0.3075
35	-0.5052	-0.3299
40	-0.4688	-0.3368
44	-0.4636	-0.3420
50	-0.4411	-0.3299
55	-0.4014	-0.3242
60	-0.3546	-0.3087
64	-0.3070	-0.2726
70	-0.2586	-0.2355
75	-0.2012	-0.1899
80	-0.1297	-0.1220
85	-0.0526	-0.0582
90	0.0407	0.0273
95	0.1444	0.1213

	UPPER	LOWER	TOTAL
CN	0.3565	-0.1982	0.1583
CC	-0.0200	0.0060	-0.0140
CM	-0.1170	0.0853	-0.0316

AIRFOIL PERFORMANCE

CL	CD	CM
0.1587	-0.0084	-0.0316

TABLE 2.18

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 122

ALPHA = 0.0

MACH NO. = 0.6979

WING DATA FILE NAME = \*WING3.DAT  
INPUT FILE NO. - 6

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.0777	0.0777
1	0.2248	-0.0694
2	-0.0856	-0.3824
5	-0.2498	-0.4625
7	-0.3072	-0.4925
9	-0.3232	-0.5228
15	-0.3427	-0.4889
20	-0.3498	-0.4871
25	-0.3481	-0.4800
29	-0.3575	-0.4603
35	-0.3700	-0.4603
40	-0.3593	-0.4532
44	-0.3664	-0.4461
50	-0.3611	-0.4264
55	-0.3451	-0.4015
60	-0.3184	-0.3676
64	-0.2995	-0.3053
70	-0.2656	-0.2497
75	-0.1920	-0.1970
80	-0.1205	-0.1270
85	-0.0592	-0.0592
90	0.0209	0.0354
95	0.1155	0.1371

	UPPER	LOWER	TOTAL
CN	0.2405	-0.3136	-0.0730
CC	-0.0007	-0.0111	-0.0118
CM	-0.0949	0.1111	0.0162

AIRFOIL PERFORMANCE		
CL	CD	CM
-0.0730	-0.0118	0.0162

TABLE 2.19

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 115

ALPHA = 6.0

MACH NO. = 0.506

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. - 35

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.5358	-0.5358
1	-2.0131	0.9416
2	-1.9900	0.7047
5	-1.6983	0.4794
7	-1.3185	0.3186
9	-1.0399	0.2404
15	-0.8704	0.1303
20	-0.7727	0.0695
25	-0.6952	0.0087
29	-0.6452	-0.0203
35	-0.5934	-0.0579
40	-0.5444	-0.0898
44	-0.5185	-0.1072
50	-0.4753	-0.1187
55	-0.4275	-0.1398
60	-0.3841	-0.1485
64	-0.3248	-0.1344
70	-0.2640	-0.1227
75	-0.1998	-0.1133
80	-0.1238	-0.0853
85	-0.0467	-0.0596
90	0.0362	-0.0140
95	0.1075	0.0386

	UPPER	LOWER	TOTAL
CN	0.5175	0.0031	0.5206
CC	-0.0705	0.0224	-0.0480
CM	-0.1394	0.0294	-0.1100

AIRFOIL PERFORMANCE

CL	CD	CM
0.5227	0.0066	-0.1100

TABLE 2.20

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 112

ALPHA = 4.0

MACH NO. = 0.507

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. = 32

ZCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.1911	-0.1911
1	-1.0975	0.7154
2	-1.2214	0.4315
5	-1.0342	0.2317
7	-0.9193	0.1098
9	-0.8274	0.0462
15	-0.6665	-0.0289
20	-0.5832	-0.0722
25	-0.5430	-0.1098
29	-0.5085	-0.1267
35	-0.4855	-0.1527
40	-0.4539	-0.1700
44	-0.4367	-0.1786
50	-0.4108	-0.1844
55	-0.3852	-0.1906
60	-0.3447	-0.1906
64	-0.2936	-0.1713
70	-0.2447	-0.1515
75	-0.1888	-0.1317
80	-0.1212	-0.0956
85	-0.0513	-0.0594
90	0.0326	0.0012
95	0.1212	0.0688

	UPPER	LOWER	TOTAL
CN	0.3961	-0.0709	0.3252
CC	-0.0416	0.0157	-0.0259
CM	-0.1166	0.0473	-0.0692

AIRFOIL PERFORMANCE

CL	CD	CM
0.3262	-0.0031	-0.0692

TABLE 2.21

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 91

ALPHA = 2.0

MACH NO. = 0.508

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. - 11

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.0198	-0.0198
1	-0.3922	0.3525
2	-0.5984	0.0693
5	-0.6014	-0.0607
7	-0.5677	-0.1387
9	-0.5302	-0.1762
15	-0.4697	-0.2023
20	-0.4466	-0.2225
25	-0.4207	-0.2427
29	-0.4005	-0.2363
35	-0.3861	-0.2536
40	-0.3631	-0.2536
44	-0.3631	-0.2565
50	-0.3515	-0.2507
55	-0.3236	-0.2485
60	-0.3034	-0.2369
64	-0.2562	-0.2092
70	-0.2162	-0.1821
75	-0.1716	-0.1481
80	-0.1152	-0.0964
85	-0.0529	-0.0458
90	0.0259	0.0247
95	0.1210	0.1046

	UPPER	LOWER	TOTAL
CN	0.2903	-0.1540	0.1363
CC	-0.0195	0.0041	-0.0155
CM	-0.0950	0.0652	-0.0298

AIRFOIL PERFORMANCE

CL	CD	CM
0.1368	-0.0107	-0.0298

TABLE 2.22

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 109

ALPHA = 2.0

MACH NO. = 0.504

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. = 29

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.0092	-0.0092
1	-0.3773	0.3589
2	-0.5862	0.0700
5	-0.5833	-0.0584
7	-0.5675	-0.1372
9	-0.5238	-0.1722
15	-0.4656	-0.1984
20	-0.4394	-0.2159
25	-0.4161	-0.2364
29	-0.4027	-0.2351
35	-0.3823	-0.2496
40	-0.3648	-0.2583
44	-0.3618	-0.2525
50	-0.3502	-0.2496
55	-0.3239	-0.2445
60	-0.2947	-0.2328
64	-0.2559	-0.2054
70	-0.2160	-0.1749
75	-0.1691	-0.1467
80	-0.1151	-0.0986
85	-0.0552	-0.0481
90	0.0235	0.0247
95	0.1174	0.1033

	UPPER	LOWER	TOTAL
CN	0.2880	-0.1517	0.1363
CC	-0.0189	0.0043	-0.0146
CM	-0.0948	0.0645	-0.0303

AIRFOIL PERFORMANCE		
CL	CD	CM
0.1367	-0.0099	-0.0303

TABLE 2.23

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 105

ALPHA = 0.0

MACH NO. = 0.506

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. = 25

XCHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.0033	-0.0033
1	0.1670	-0.1737
2	-0.1066	-0.4170
5	-0.2275	-0.4372
7	-0.2736	-0.4274
9	-0.2851	-0.4332
15	-0.2938	-0.3956
20	-0.2909	-0.3870
25	-0.2909	-0.3870
29	-0.3003	-0.3746
35	-0.3032	-0.3746
40	-0.3003	-0.3629
44	-0.3061	-0.3571
50	-0.2974	-0.3339
55	-0.2838	-0.3223
60	-0.2722	-0.3049
64	-0.2458	-0.2575
70	-0.2202	-0.2179
75	-0.1724	-0.1748
80	-0.1142	-0.1188
85	-0.0606	-0.0594
90	0.0105	0.0210
95	0.0944	0.1118

	UPPER	LOWER	TOTAL
CN	0.2057	-0.2631	-0.0574
CC	-0.0017	-0.0120	-0.0137
CM	-0.0811	0.0928	0.0117

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0574	-0.0137	0.0117

TABLE 2.24

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 89

ALPHA = 2.0

MACH NO. = 0.3063

WING DATA FILE NAME = \*WING2.DAT  
INPUT FILE NO. = 9

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	-0.0320	-0.0320
1	-0.3981	0.3341
2	-0.5758	0.0640
5	-0.5545	-0.0498
7	-0.5154	-0.1280
9	-0.4871	-0.1564
15	-0.4377	-0.1848
20	-0.3953	-0.1990
25	-0.3742	-0.2204
29	-0.3839	-0.2204
35	-0.3768	-0.2346
40	-0.3483	-0.2346
44	-0.3483	-0.2275
50	-0.3270	-0.2275
55	-0.3057	-0.2204
60	-0.2844	-0.2133
64	-0.2473	-0.1955
70	-0.2099	-0.1668
75	-0.1668	-0.1409
80	-0.1121	-0.0978
85	-0.0575	-0.0518
90	0.0144	0.0173
95	0.1035	0.0891

	UPPER	LOWER	TOTAL
CN	0.2741	-0.1417	0.1323
CC	-0.0182	0.0040	-0.0142
CM	-0.0914	0.0609	-0.0305

AIRFOIL PERFORMANCE

CL	CD	CM
0.1327	-0.0095	-0.0305

NACA SECTION ANALYSIS  
0012-64

RUN NO. = 224

ALPHA = 4.0

MACH NO. = 0.882

WING DATA FILE NAME = \*WING.DAT  
INPUT FILE NO. - 32

%CHORD	UPPER SURFACE	LOWER SURFACE
	CP LOCAL	CP LOCAL
0	0.2553	0.2553
1	-0.0807	0.5914
2	-0.4975	0.3050
5	-0.7396	0.1074
7	-0.6886	0.0385
9	-0.6833	-0.0346
15	-0.6582	-0.1196
20	-0.7335	-0.1807
25	-0.7388	-0.2498
29	-0.7775	-0.2814
35	-0.8212	-0.3133
40	-0.7417	-0.3637
44	-0.4252	-0.4288
50	-0.3311	-0.4805
55	-0.3103	-0.5202
60	-0.3050	-0.5694
64	-0.2970	-0.5978
70	-0.2933	-0.6203
75	-0.2858	-0.6428
80	-0.2676	-0.6893
85	-0.2526	-0.7129
90	-0.2280	-0.4185
95	-0.2055	-0.2269

	UPPER	LOWER	TOTAL
CN	0.4537	-0.3536	0.1001
CC	-0.0104	0.0382	0.0278
CM	-0.1751	0.2271	0.0520

## AIRFOIL PERFORMANCE

CL	CD	CM
0.0979	0.0347	0.0520

FIGURE 1

NACA 0012-64 SECTION  
PRESSURE DISTRIBUTIONS AND FORCES

FIGURE 1.1

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
184 4.0 0.890

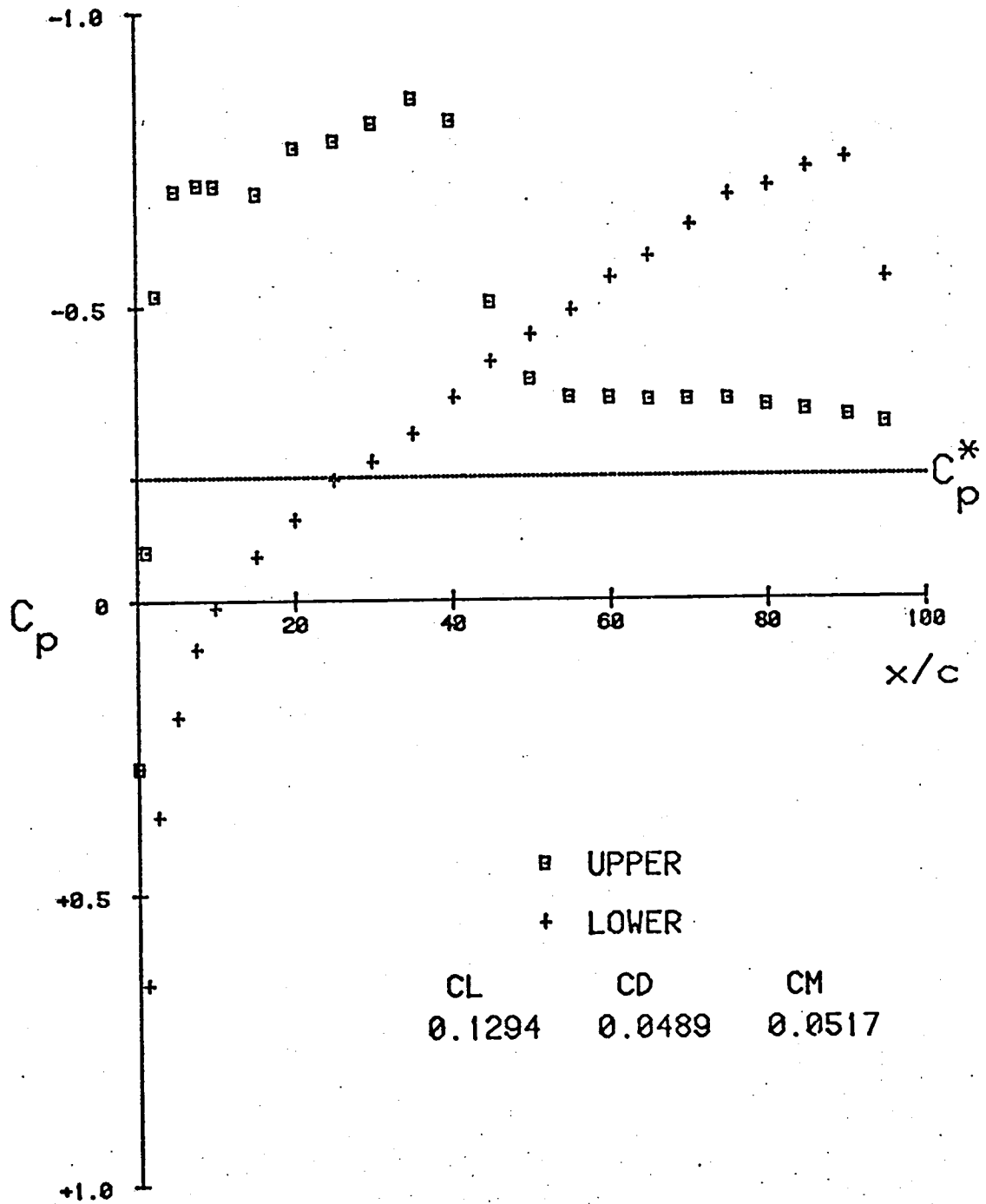


FIGURE 1.2

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
176 2.0 0.891

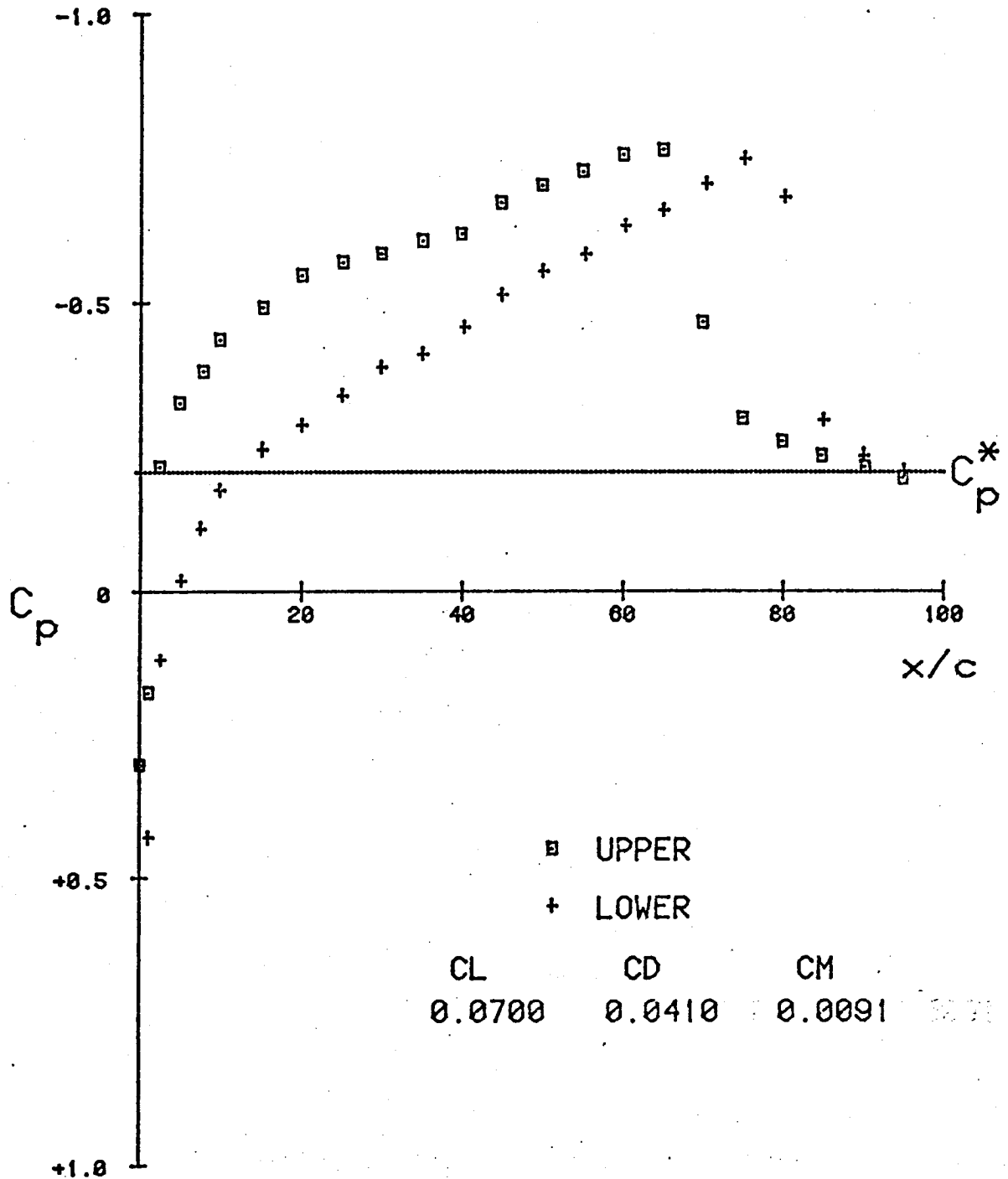


FIGURE 1.3

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
108 0.0 0.866

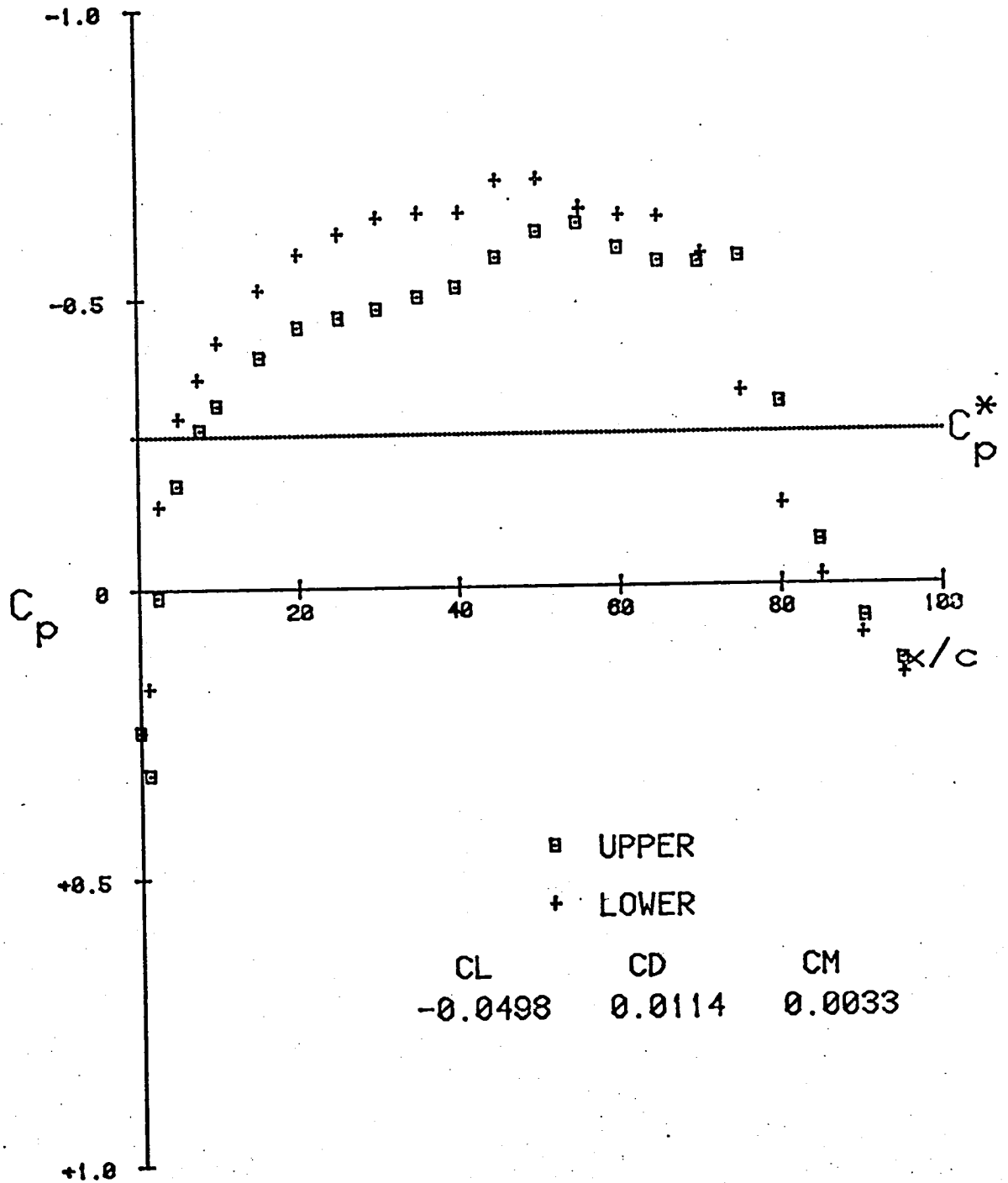


FIGURE 1.4

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
168 4.5 0.846

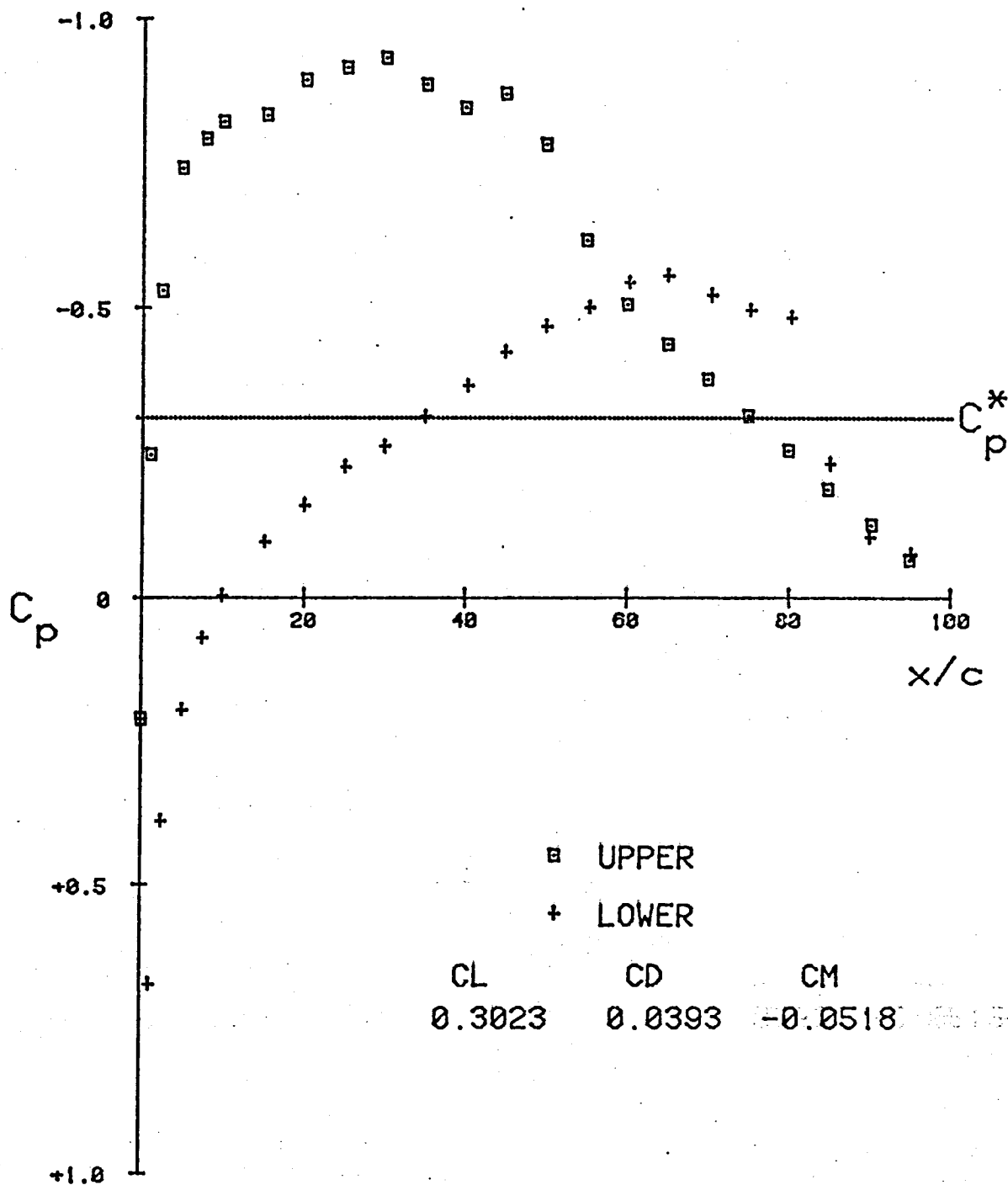


FIGURE 1.5

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
170 4.5 0.849

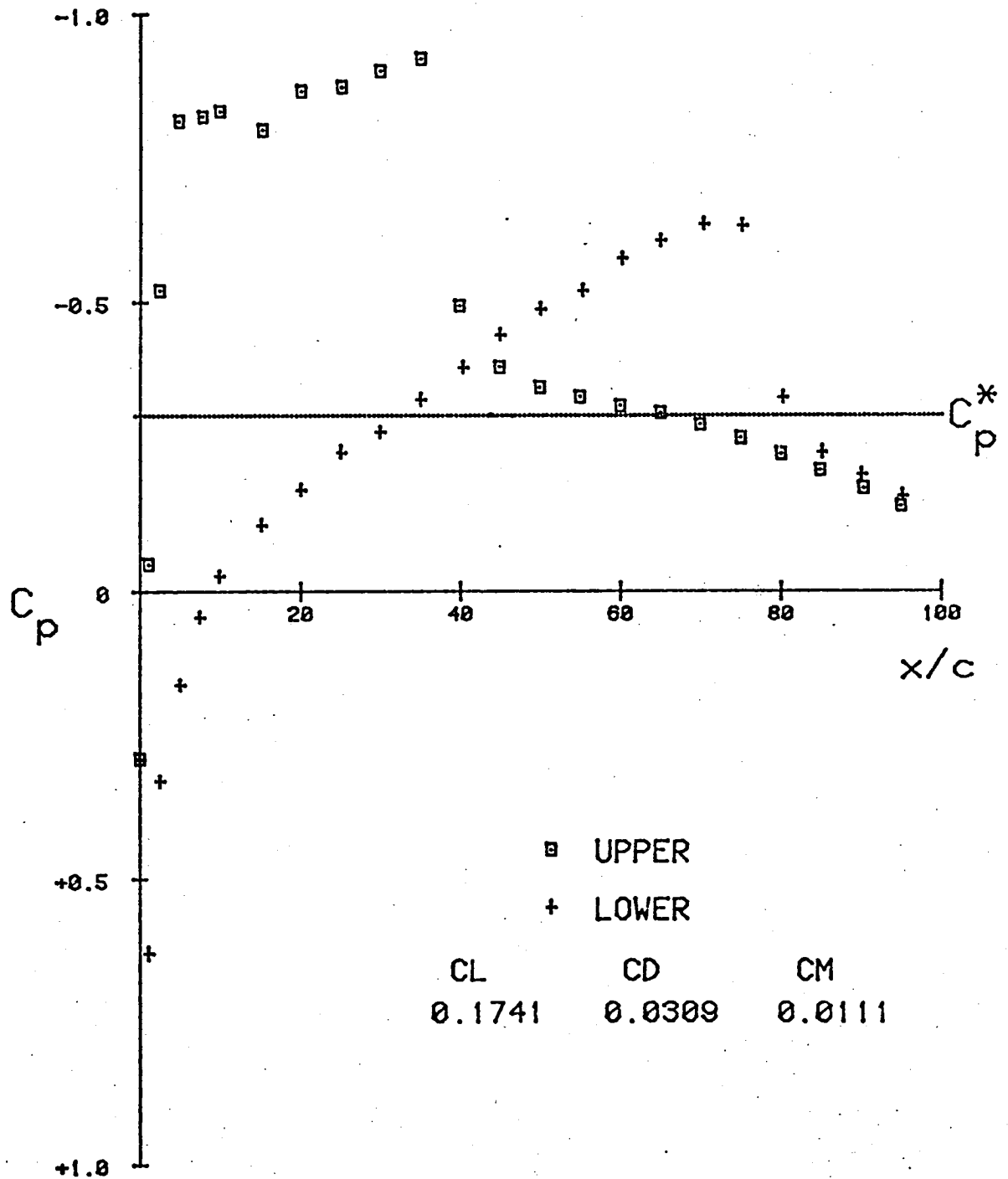


FIGURE 1.6

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
172 2.0 0.848

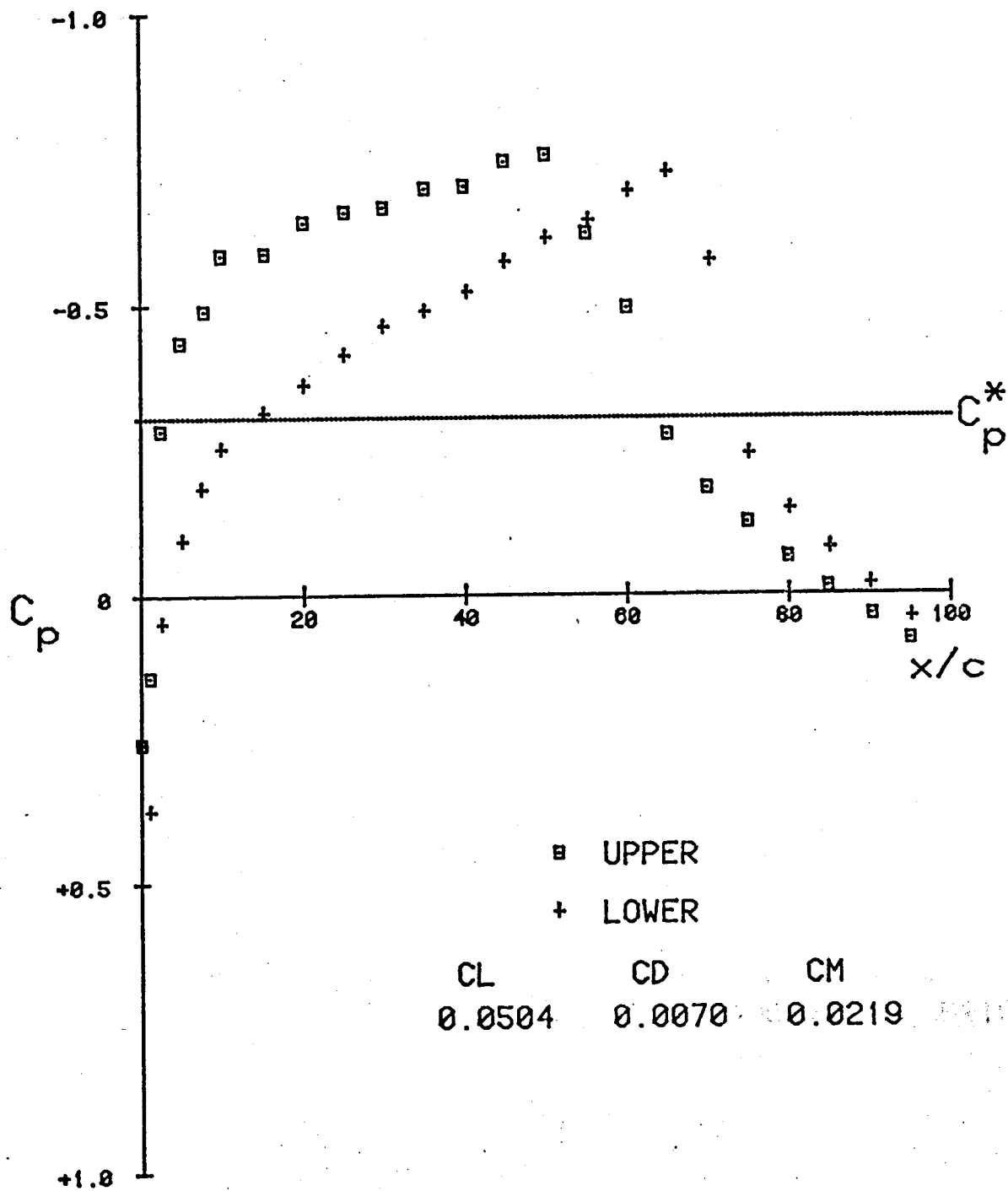


FIGURE 1.7

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
162 2.0 0.839

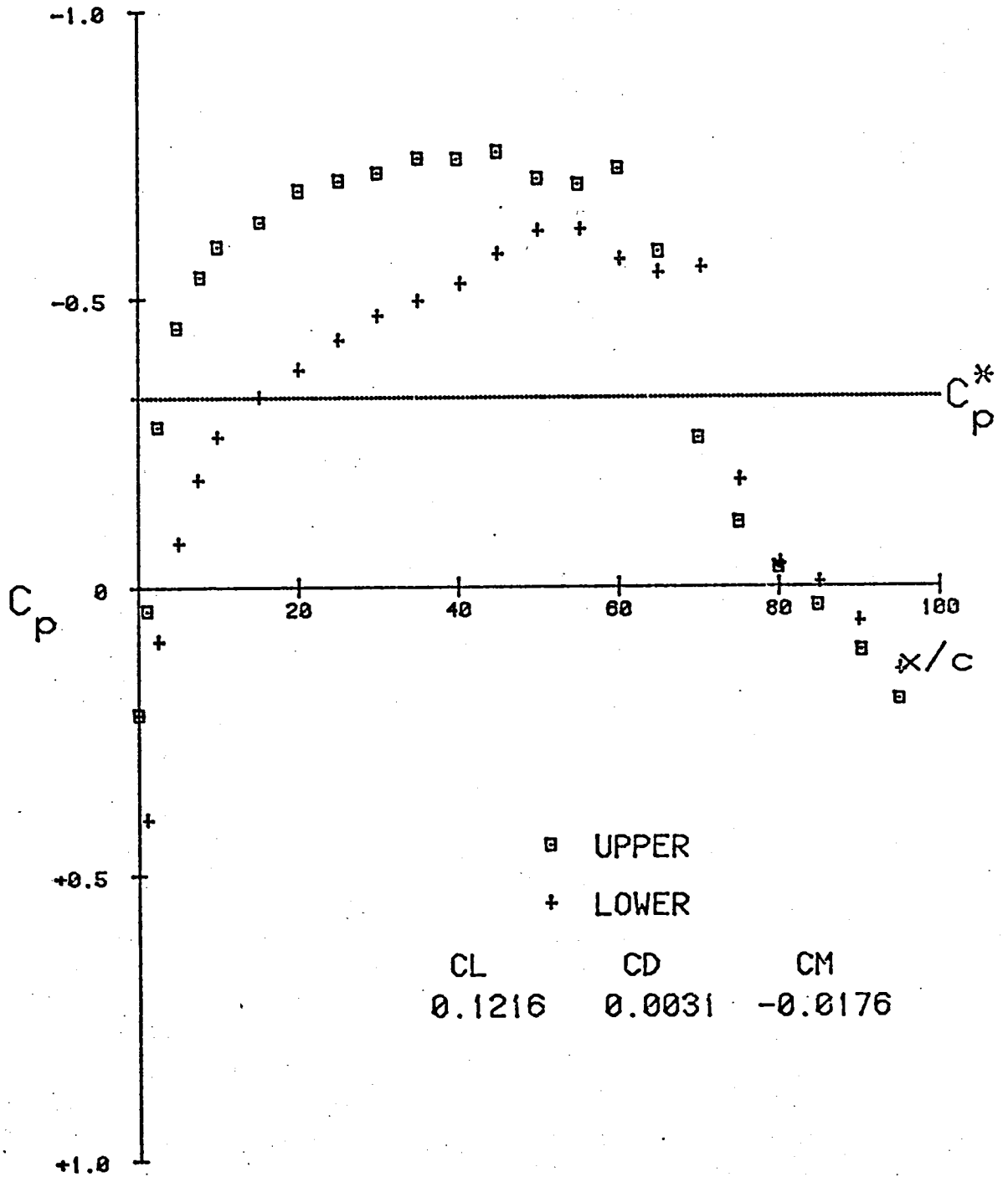


FIGURE 1.8

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
100 2.0 0.840

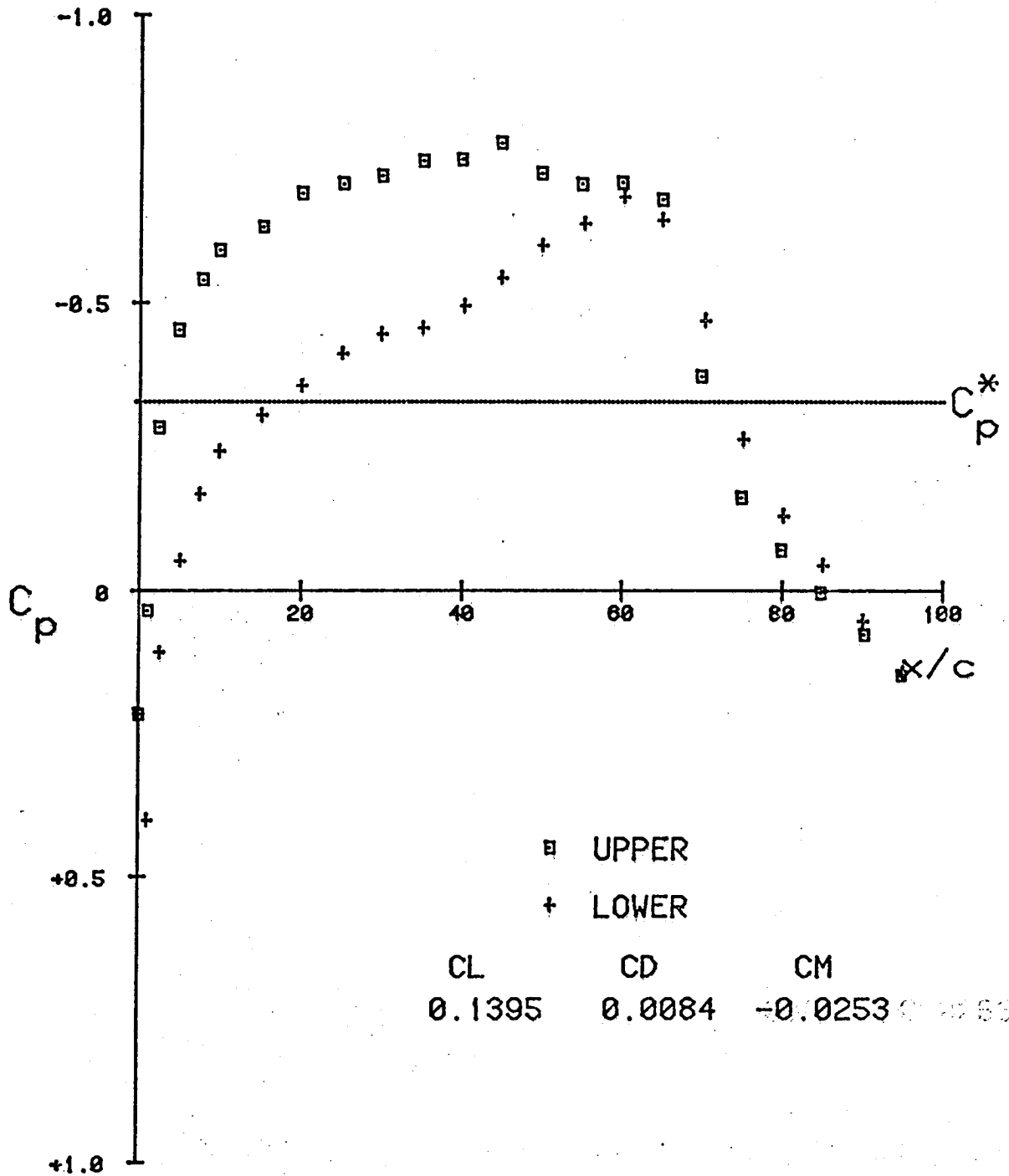


FIGURE 1.9

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
136 0.0 0.840

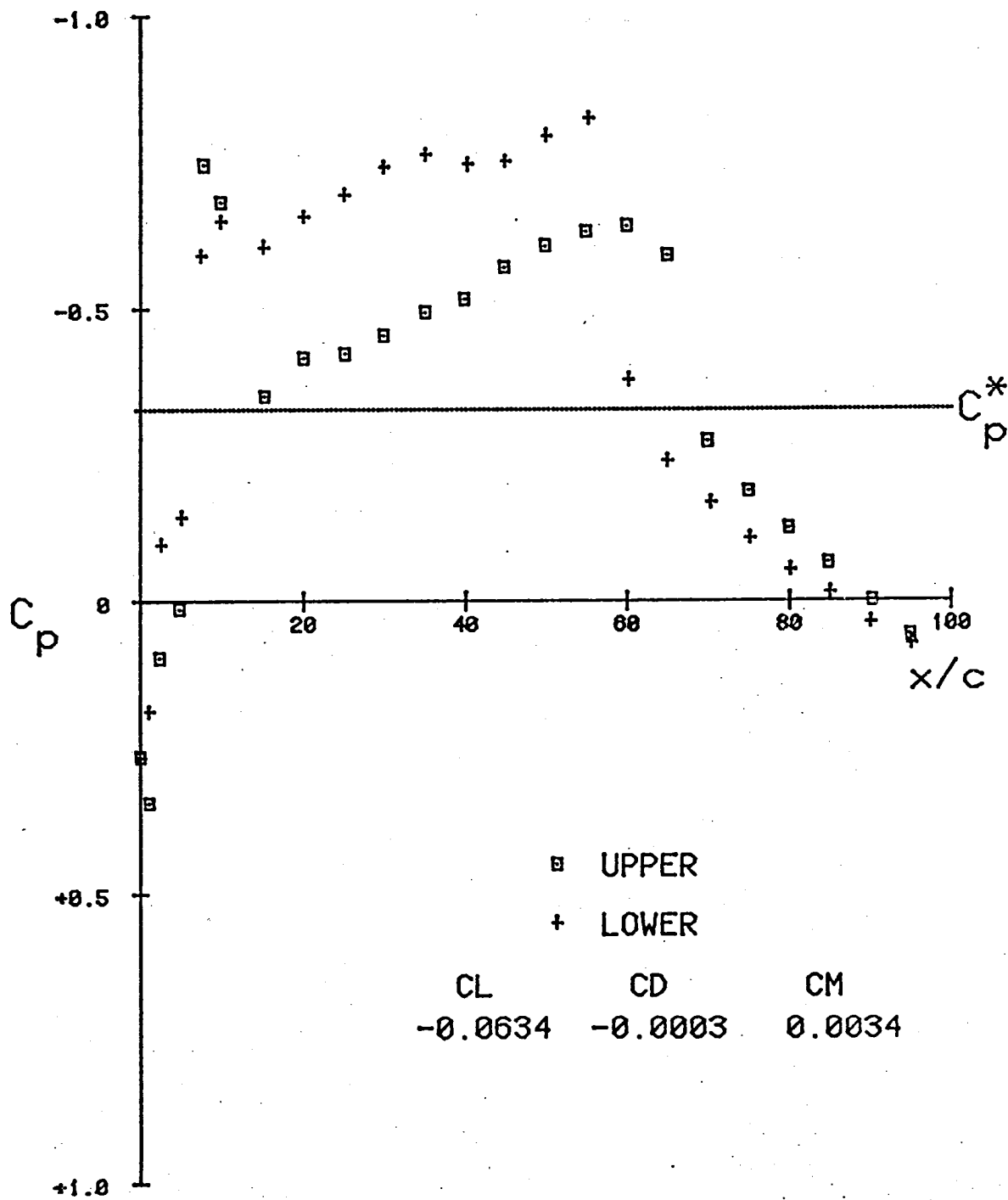


FIGURE 1.10

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
119 2.0 0.810

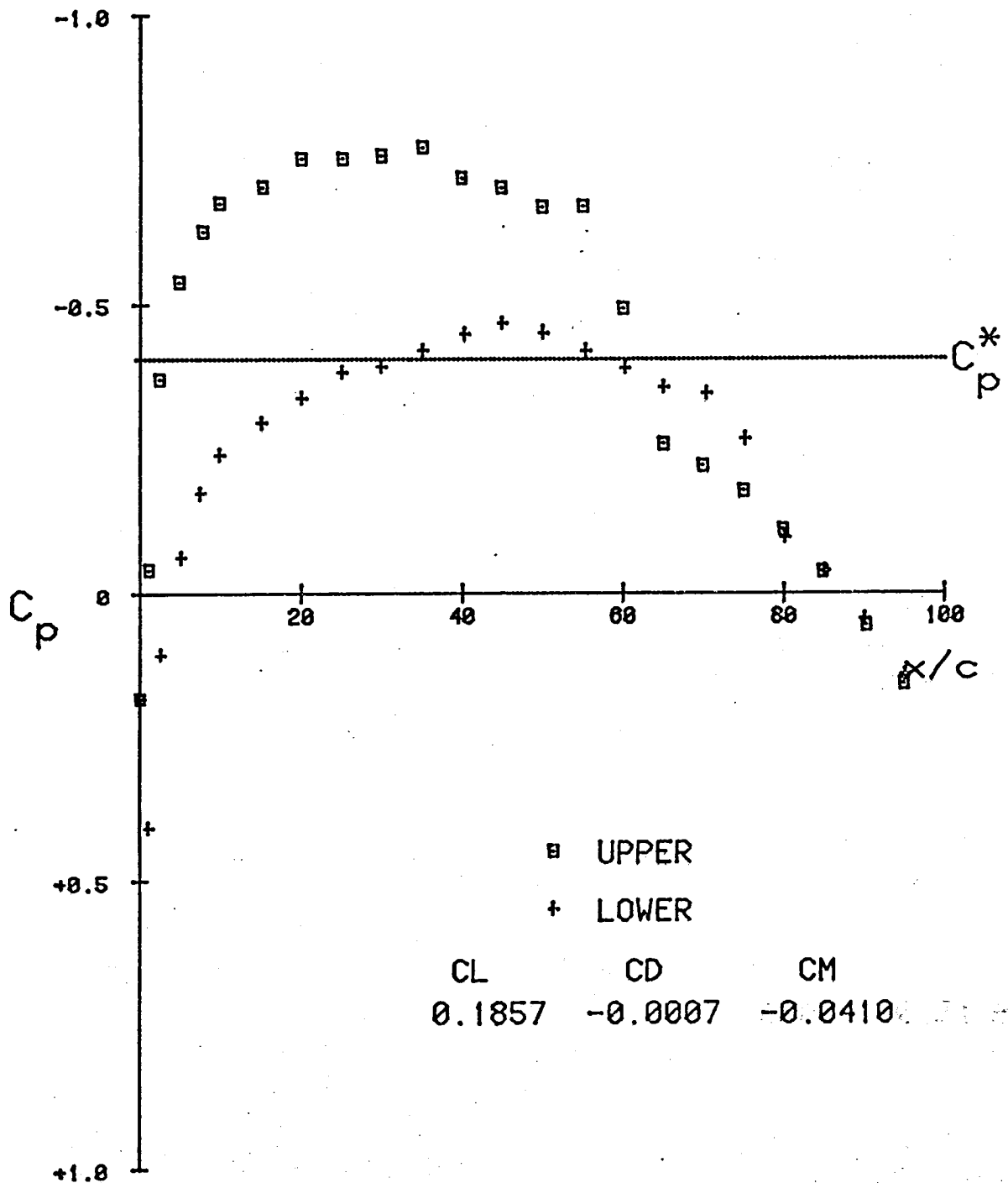


FIGURE 1.11

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
188 0.0 0.786

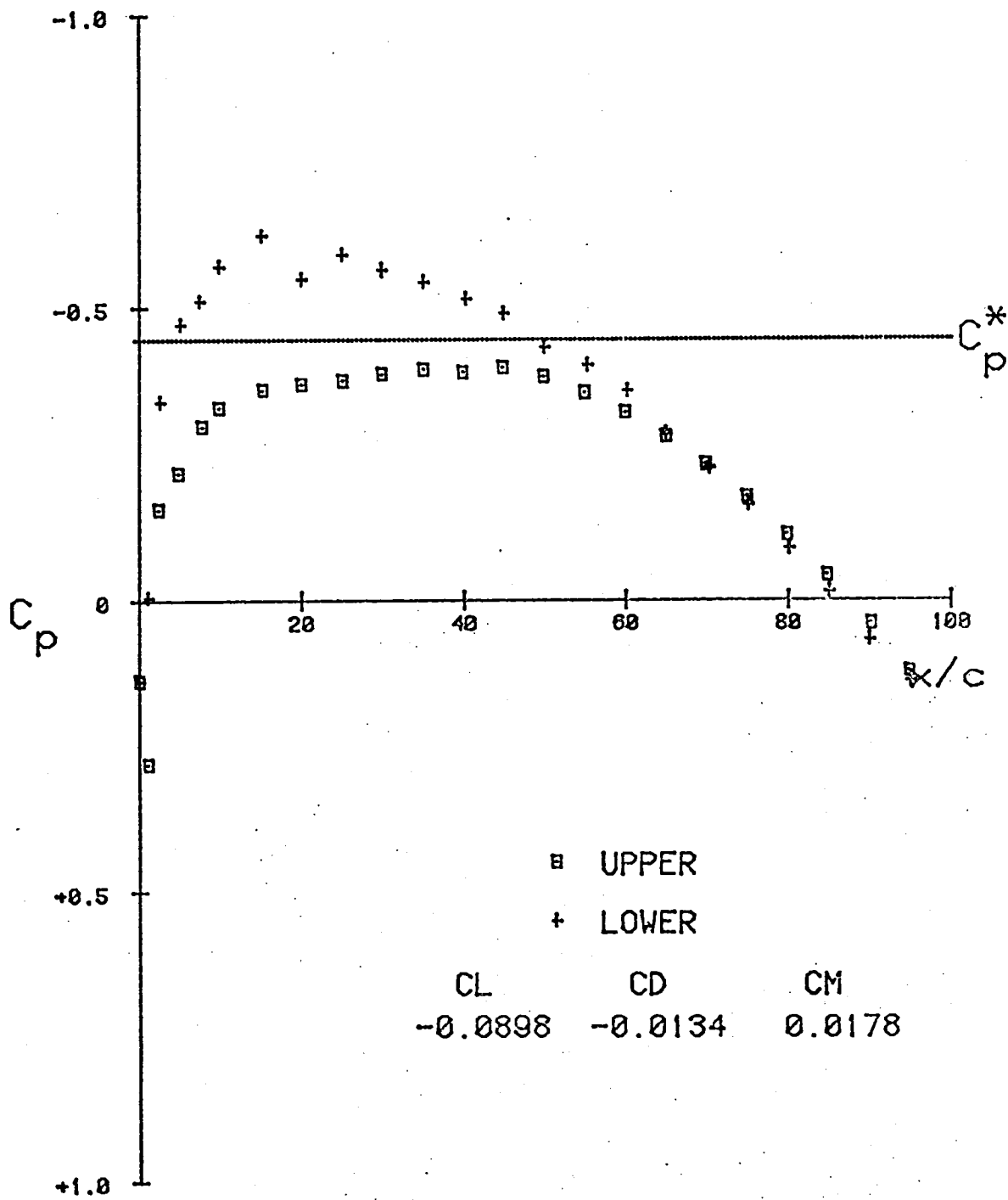


FIGURE 1.12

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
105 0.0 0.753

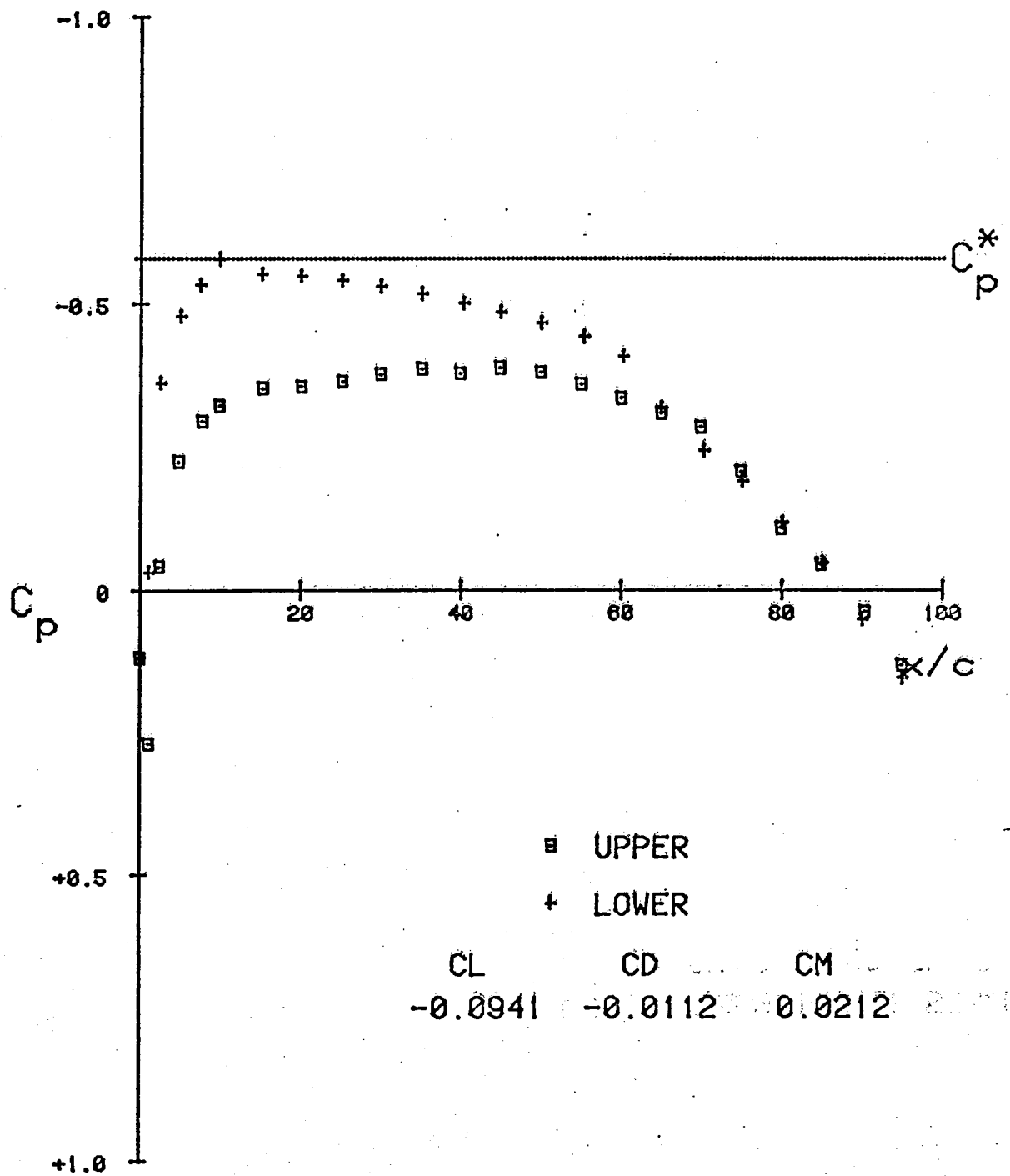


FIGURE 1.13

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
72 4.0 0.705

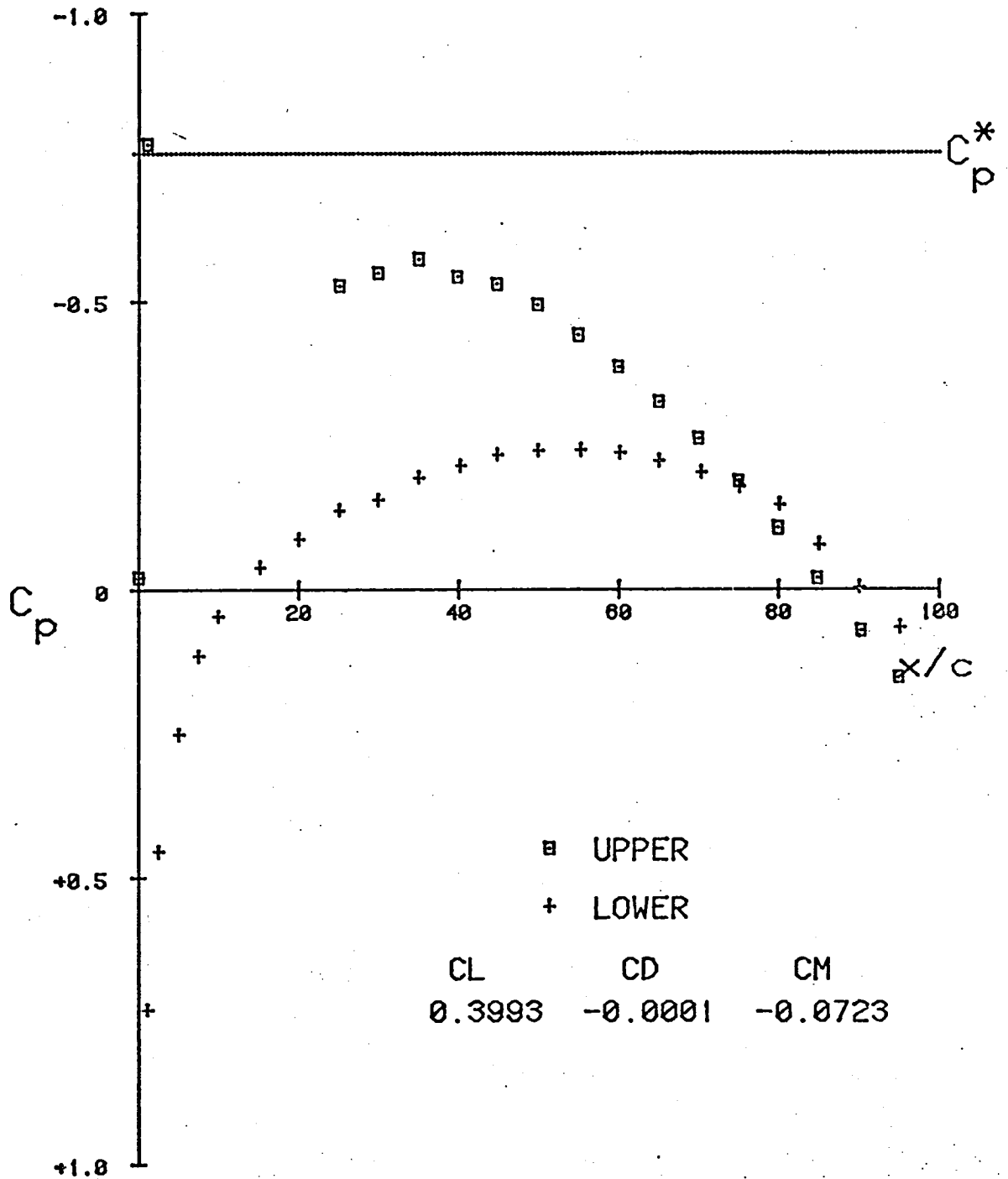


FIGURE 1.14

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
63 4.0 0.702

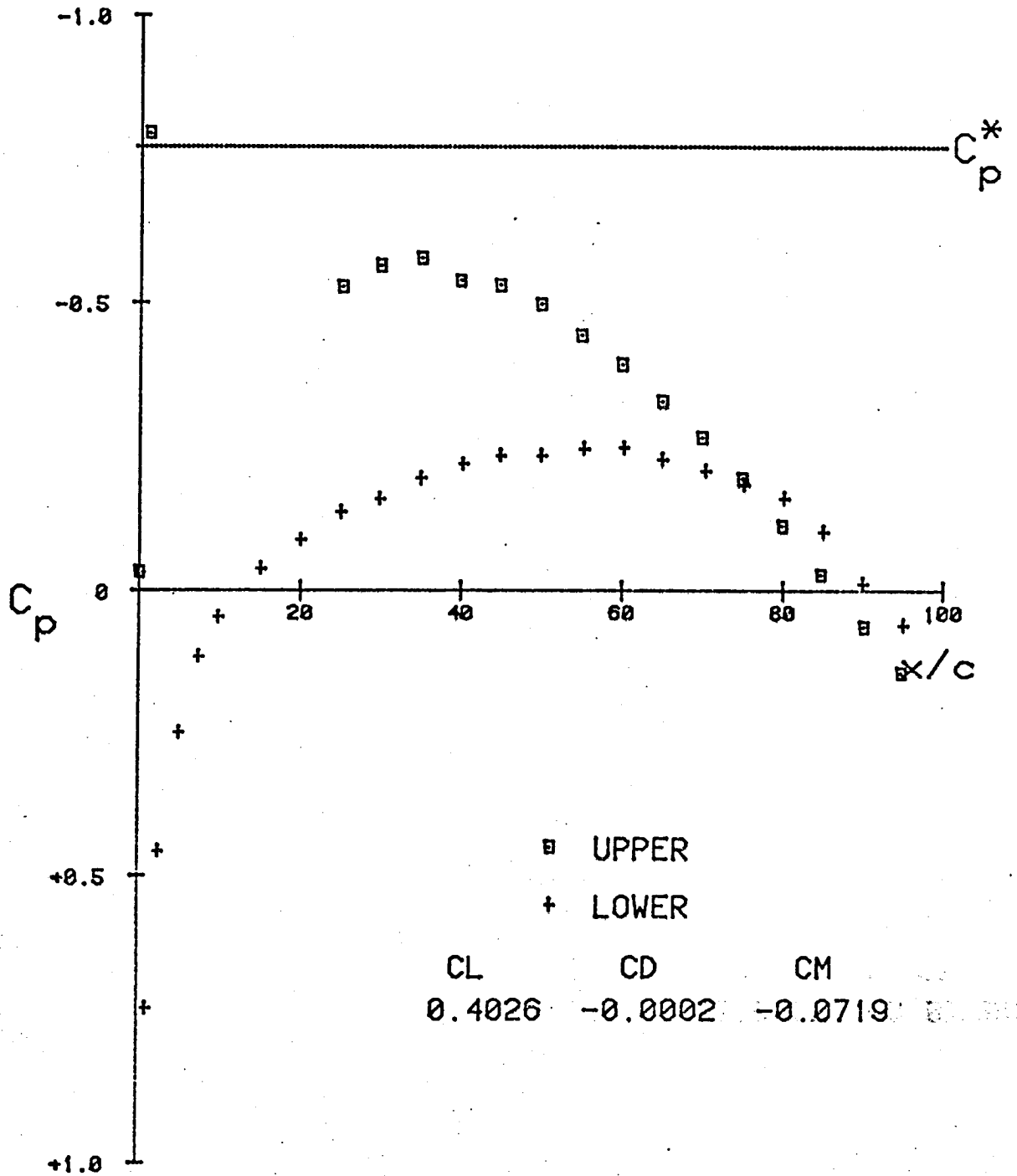


FIGURE 1.17

# NACA 0012-64 Section

RUN NO    ALPHA    MACH NO  
93        2.0        0.712

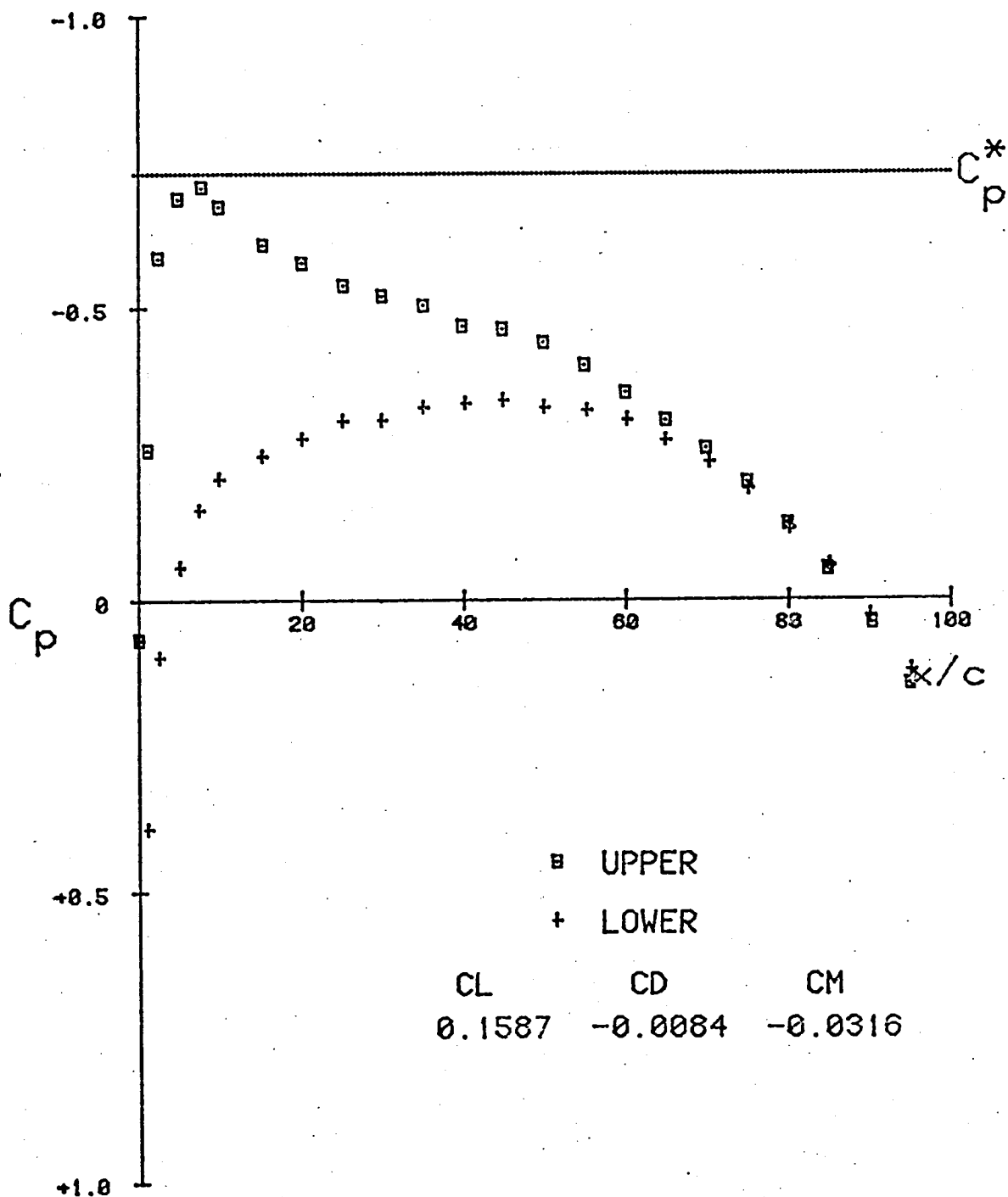
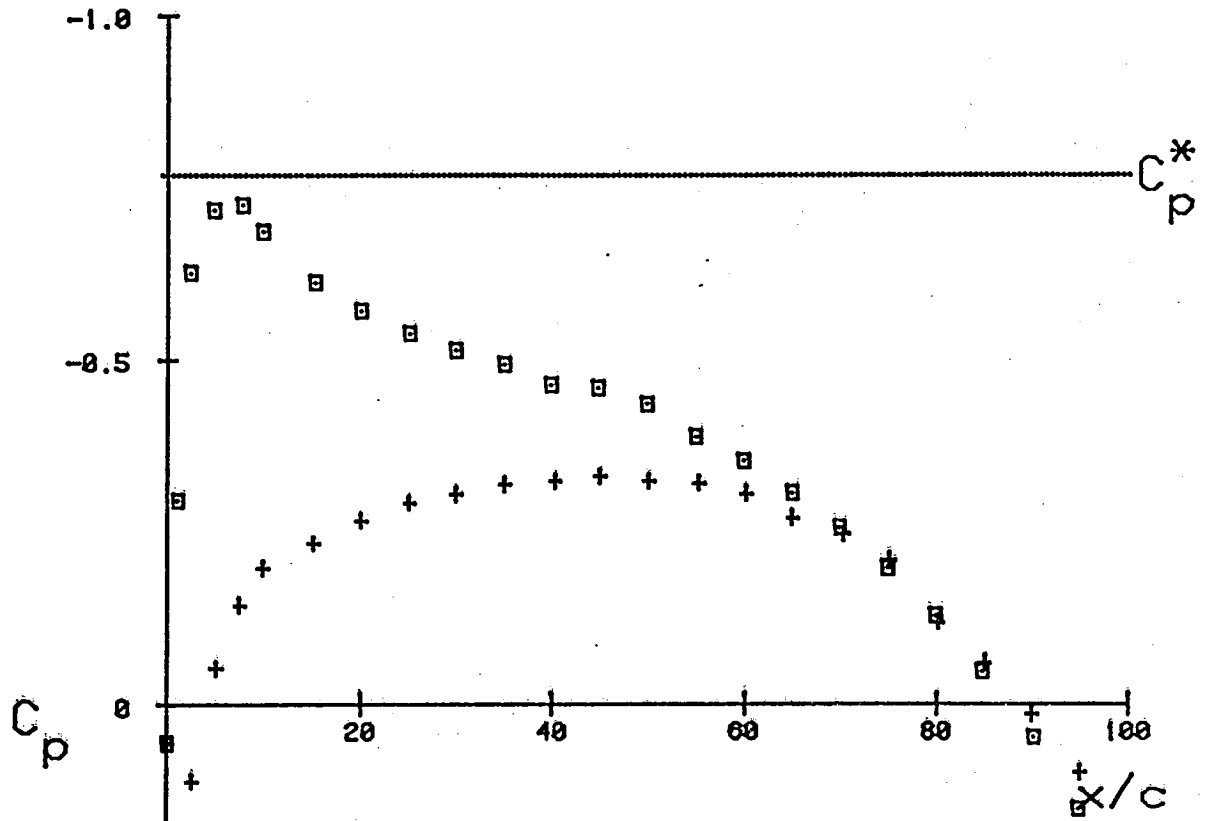


FIGURE 1.16

# NACA 0012-64 Section

RUN NO 65    ALPHA 2.0    MACH NO 0.703



□ UPPER  
+ LOWER

CL    CD    CM  
0.1600    -0.0090    -0.0283

FIGURE 1.15

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
69 3.0 0.701

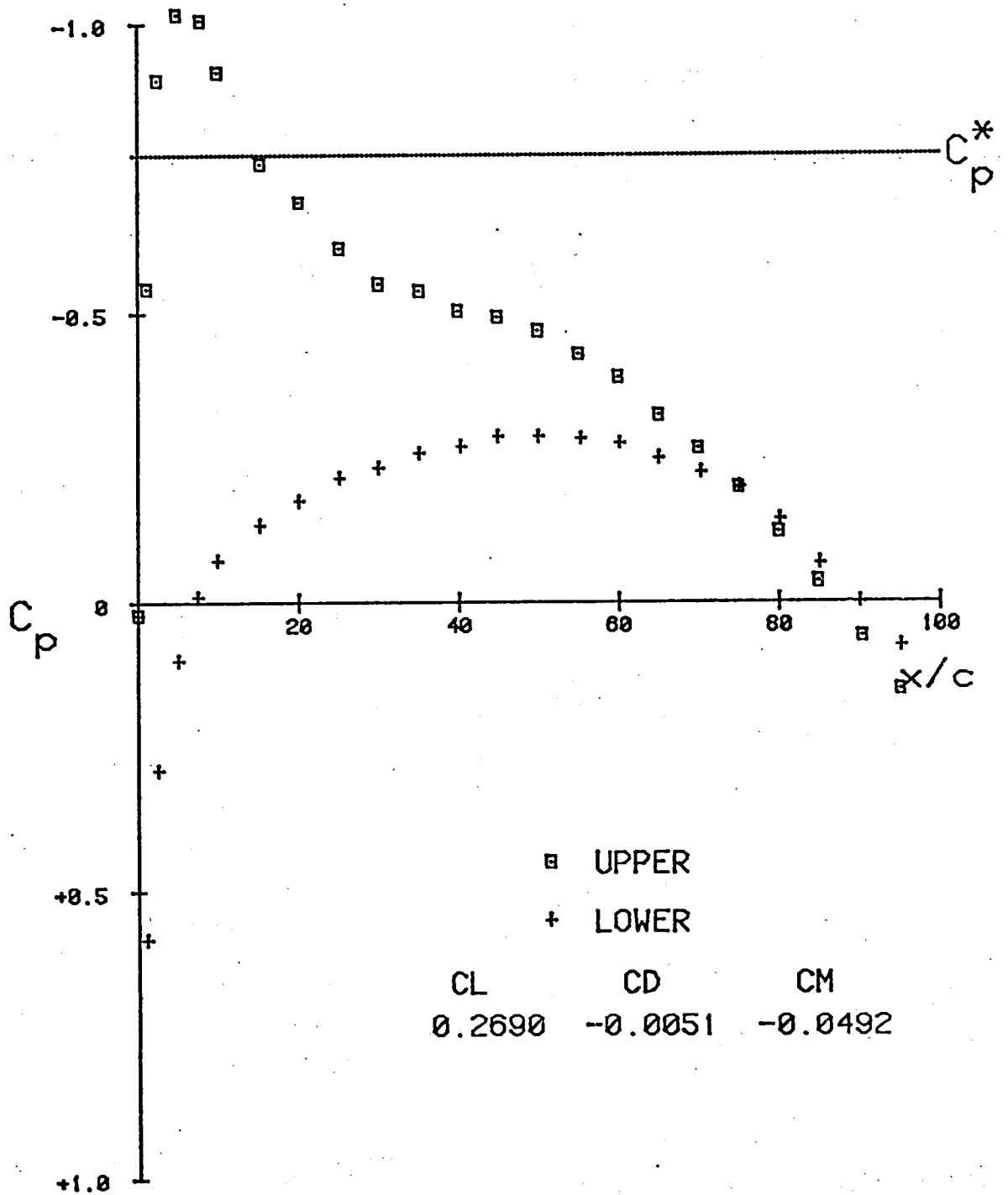


FIGURE 1.18

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
122 0.0 0.698

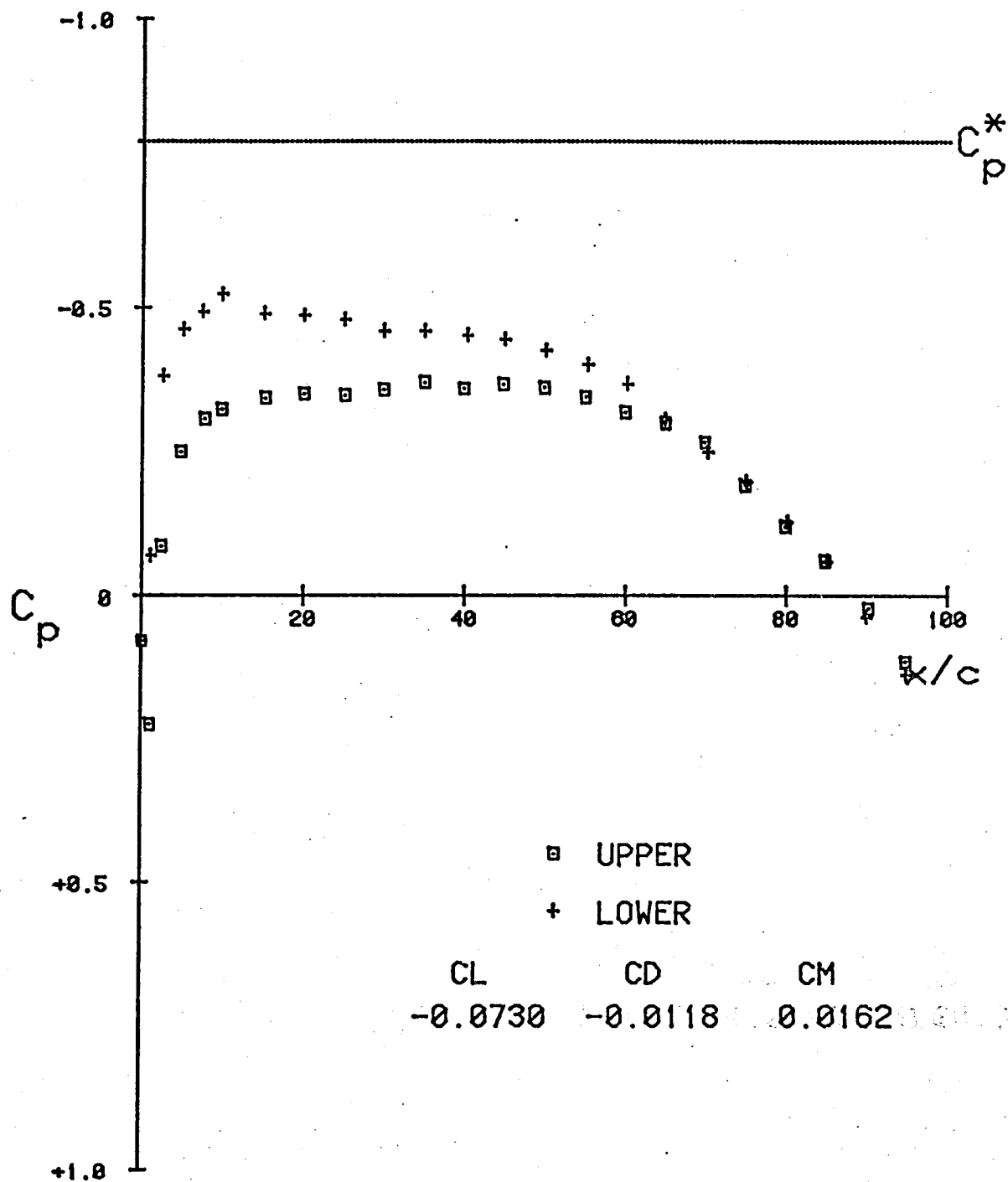


FIGURE 1.19

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
115 6.0 0.506

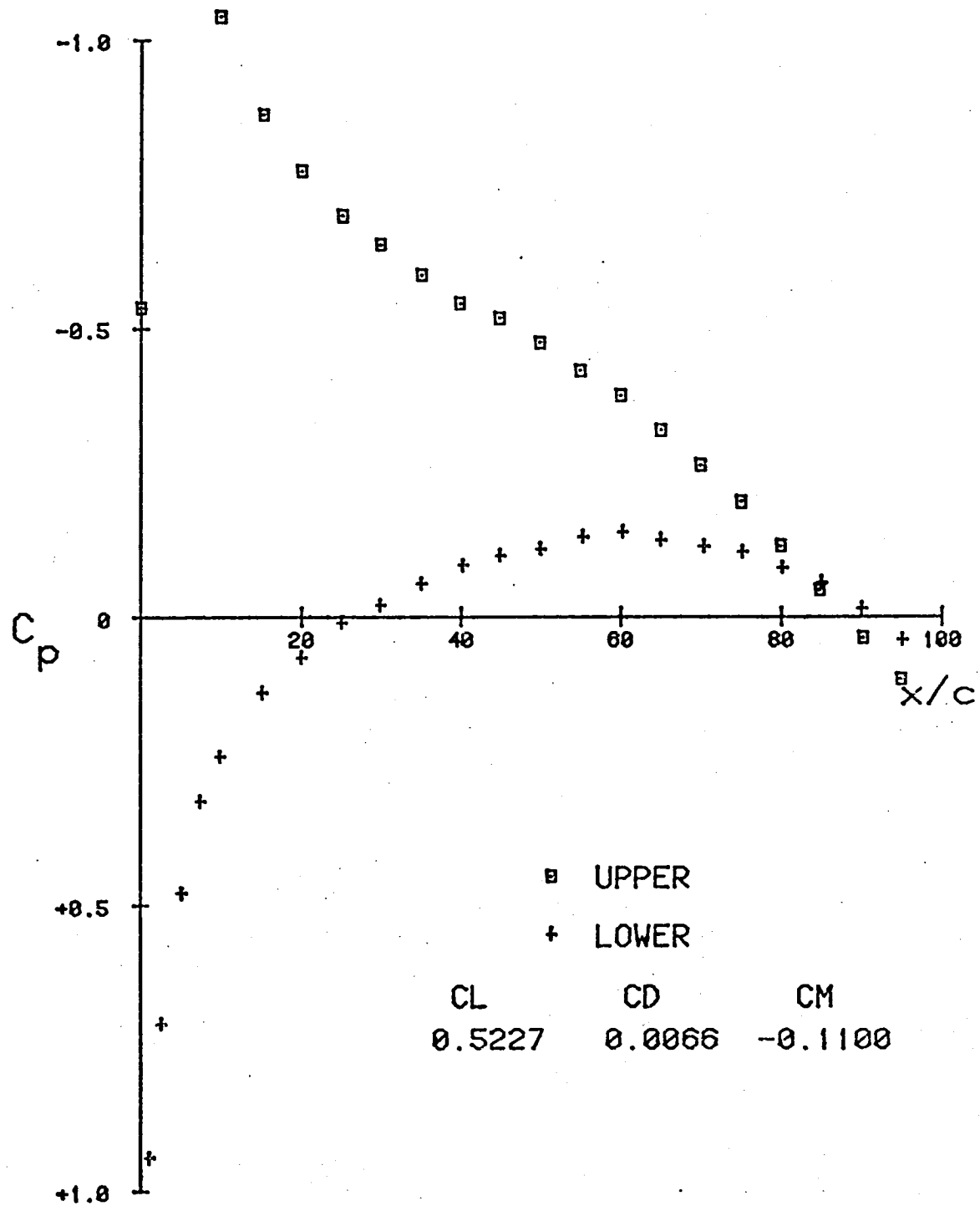


FIGURE 1.20

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
112 4.0 0.507

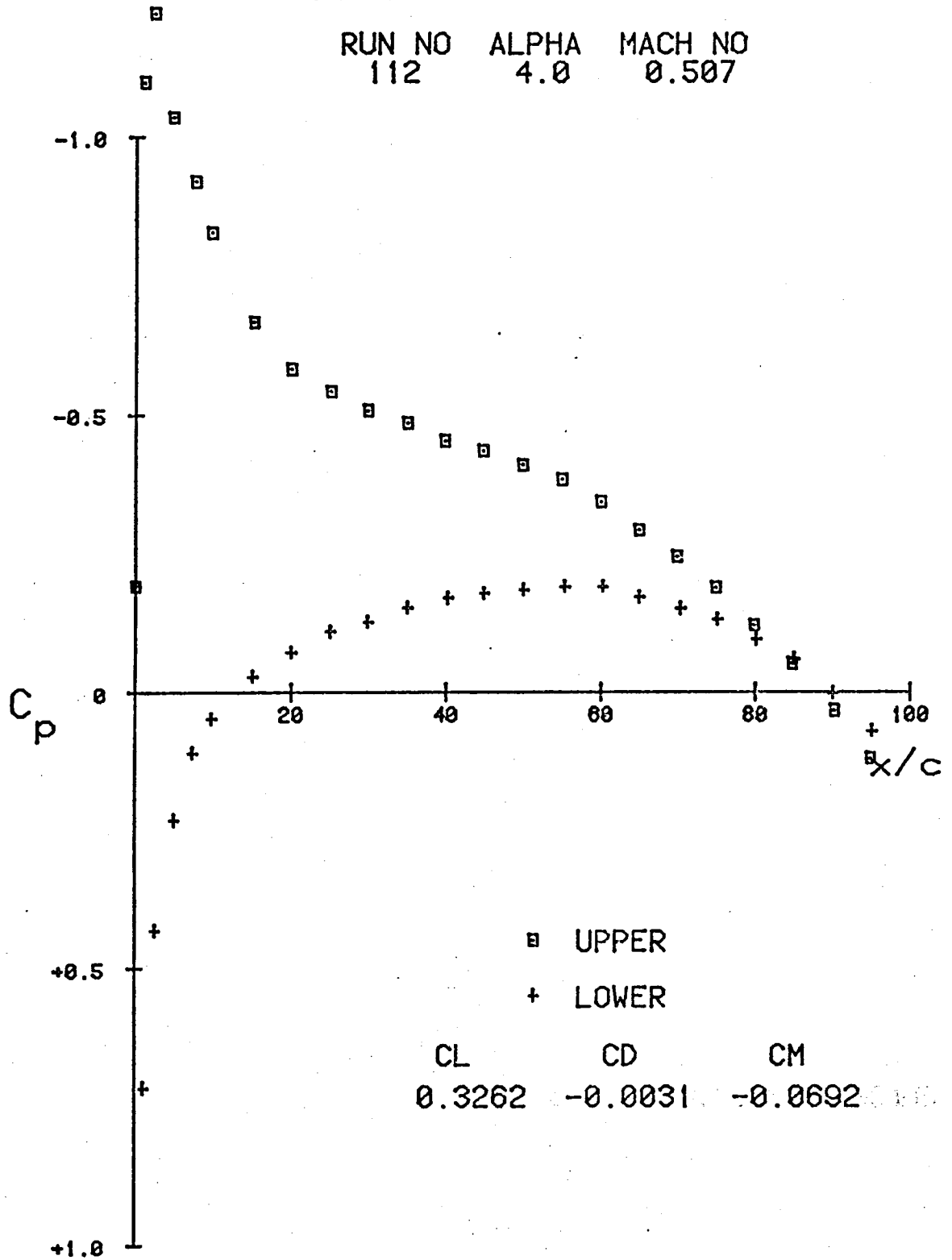


FIGURE 1.21

# NACA 0012-64 Section

RUN NO    ALPHA    MACH NO  
 91        2.0        0.508

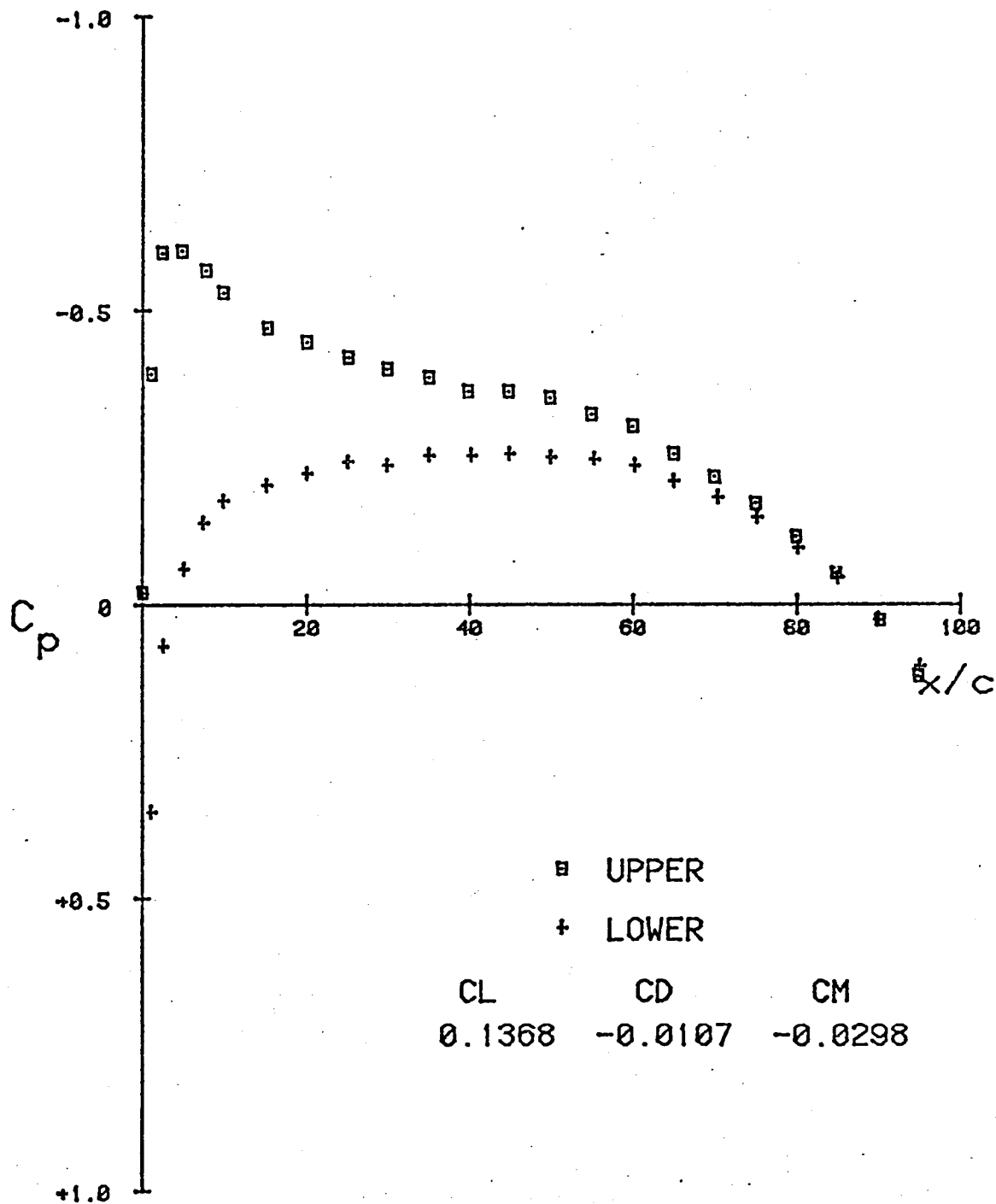


FIGURE 1.22

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
109 2.0 0.504

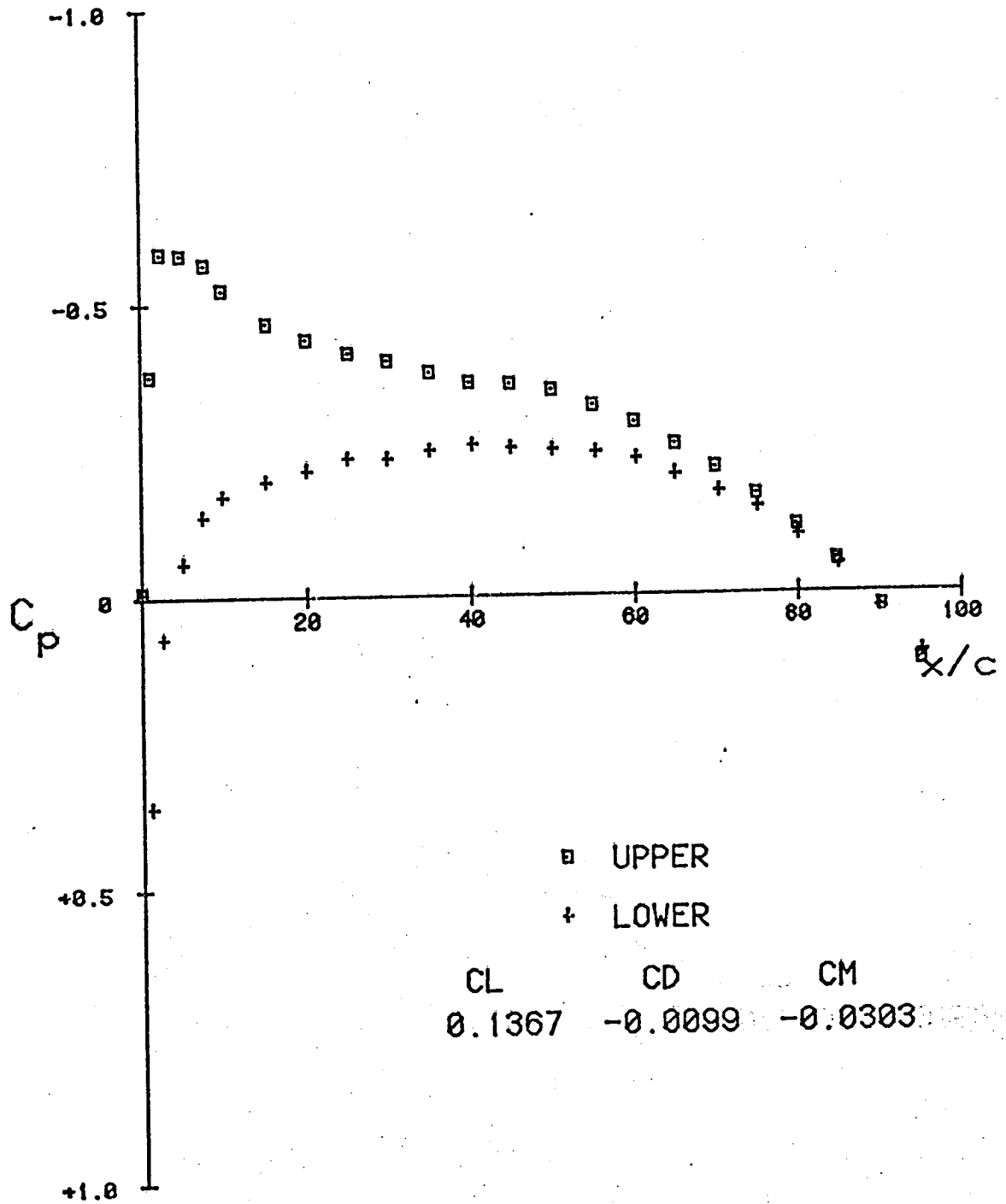


FIGURE 1.23

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
105 0.0 0.506

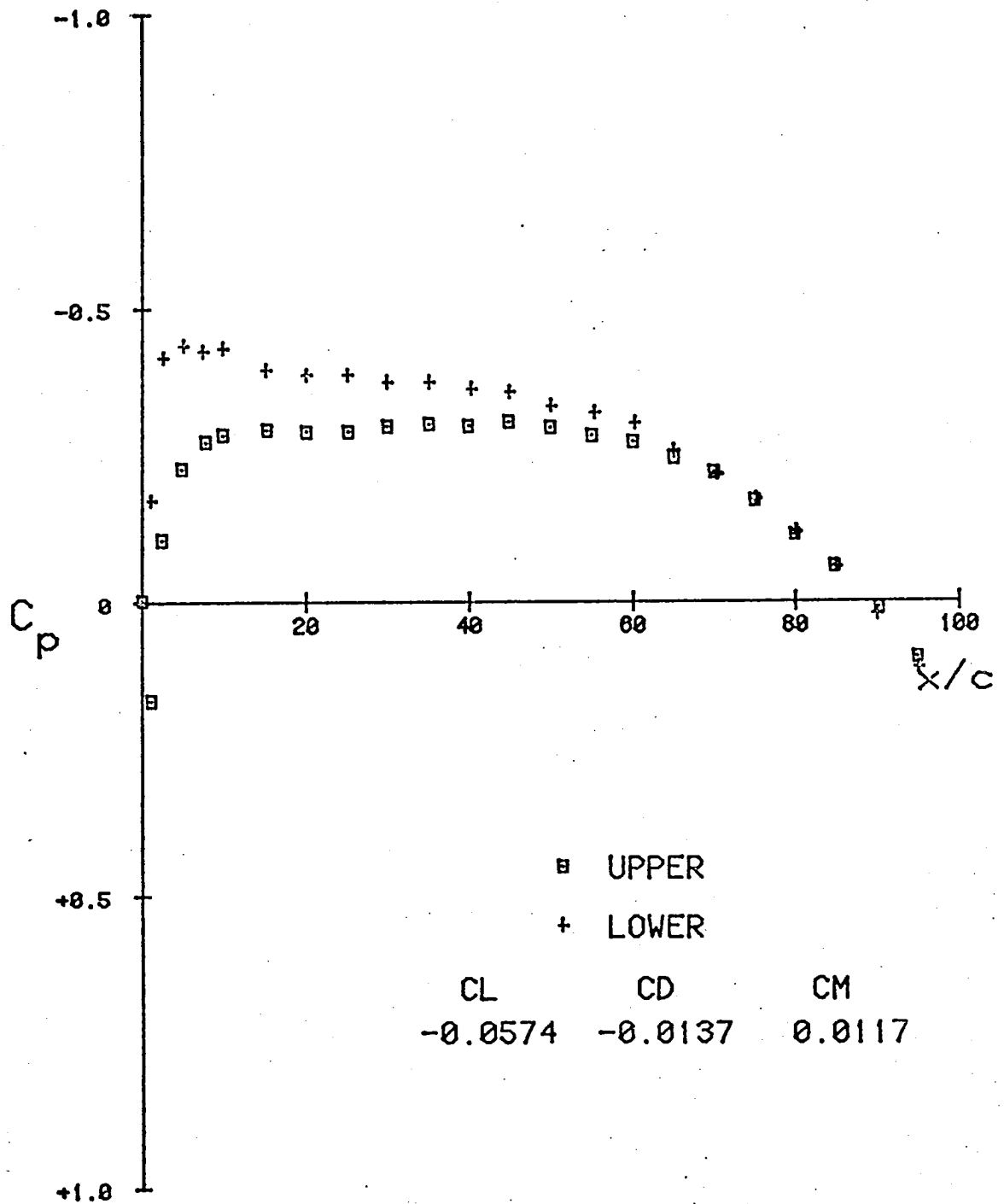


FIGURE 1.24

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
89 2.0 0.306

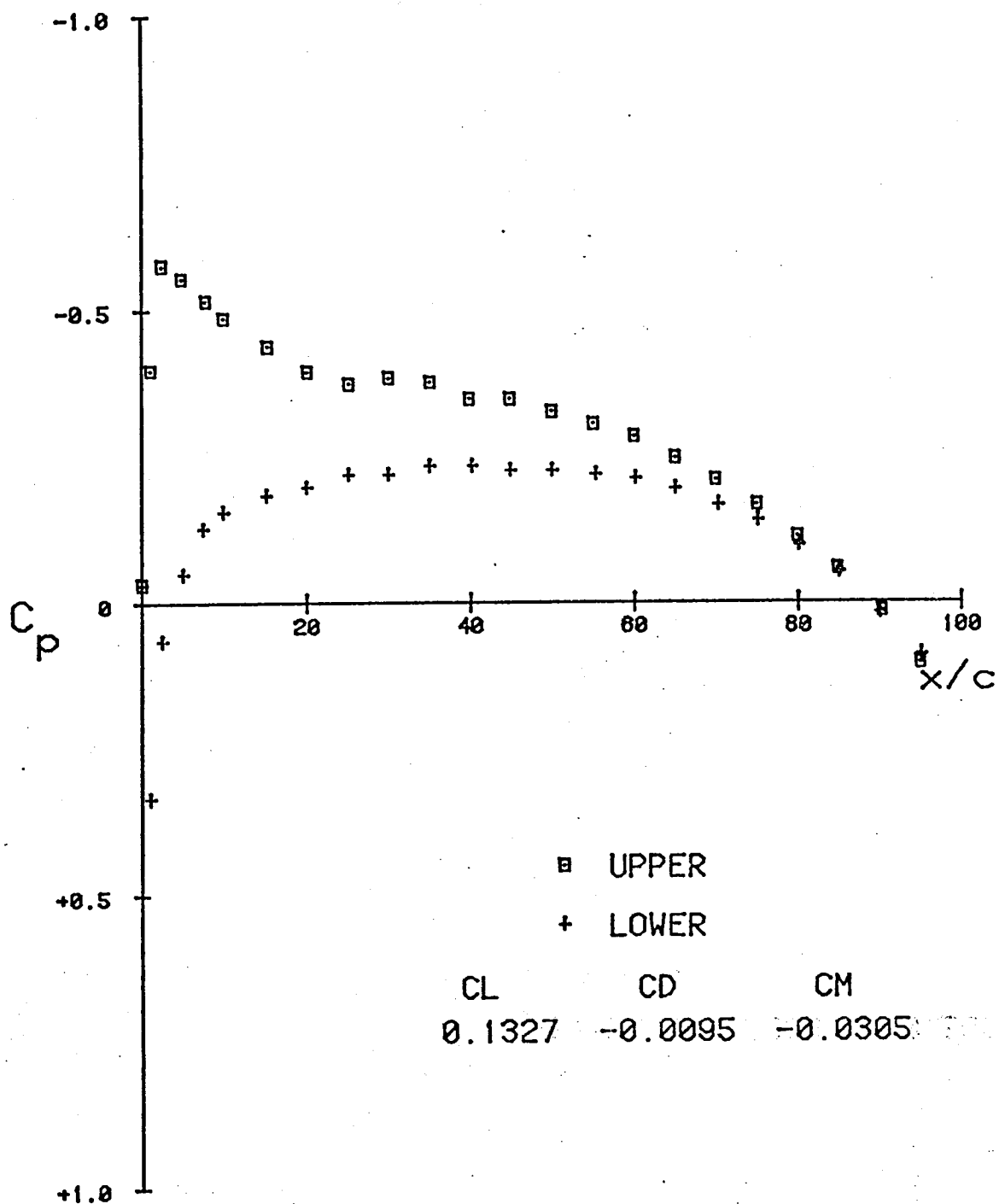


FIGURE 1.25

# NACA 0012-64 Section

RUN NO ALPHA MACH NO  
224 4.0 0.882

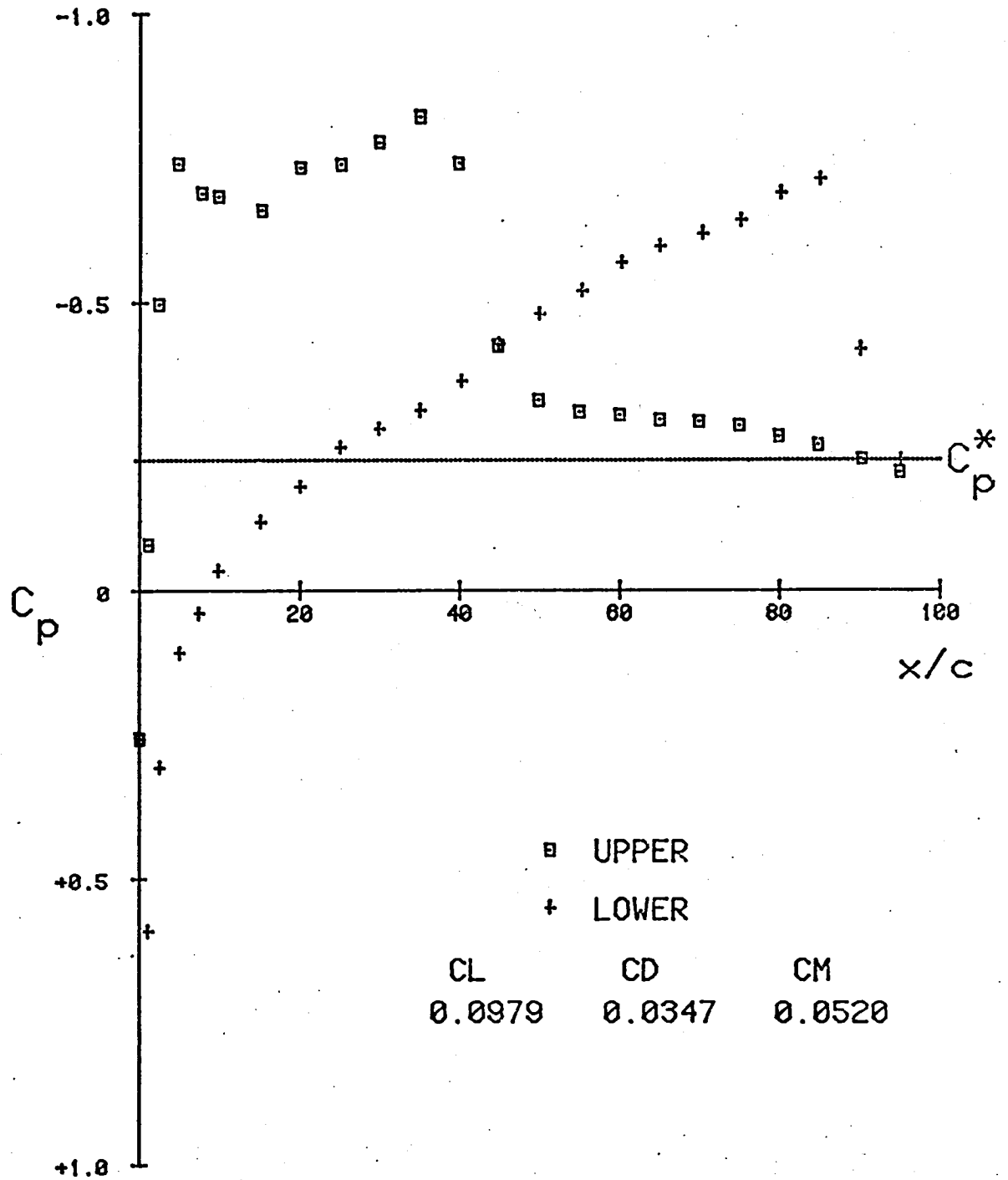


FIGURE 2

MACH NUMBER DISTRIBUTIONS ALONG

THE CENTRELINE OF EACH FLEXIBLE

WALL

FIGURE 2.1

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
184 4.0 0.886

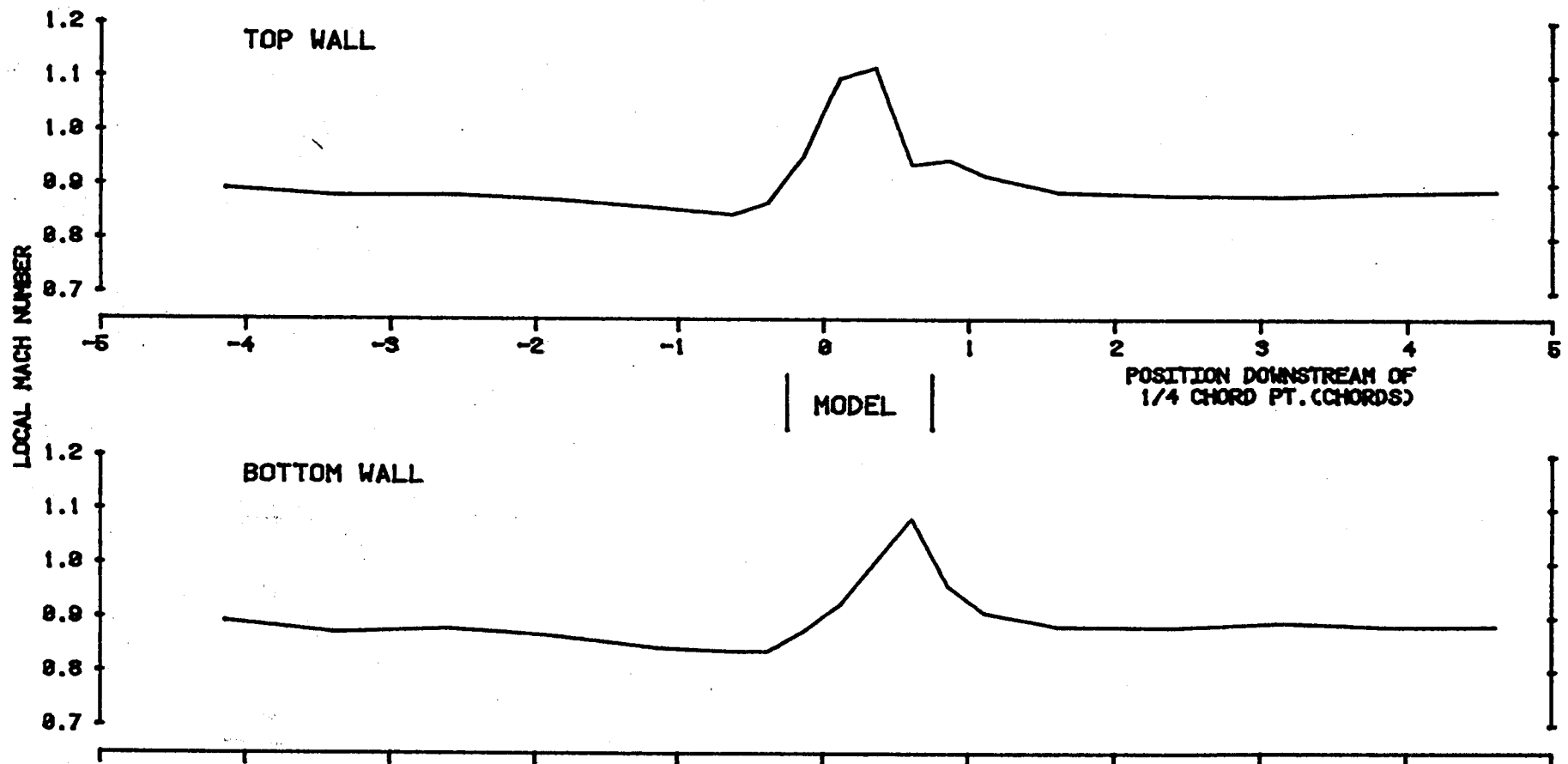


FIGURE 2.2

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
176 2.0 0.891

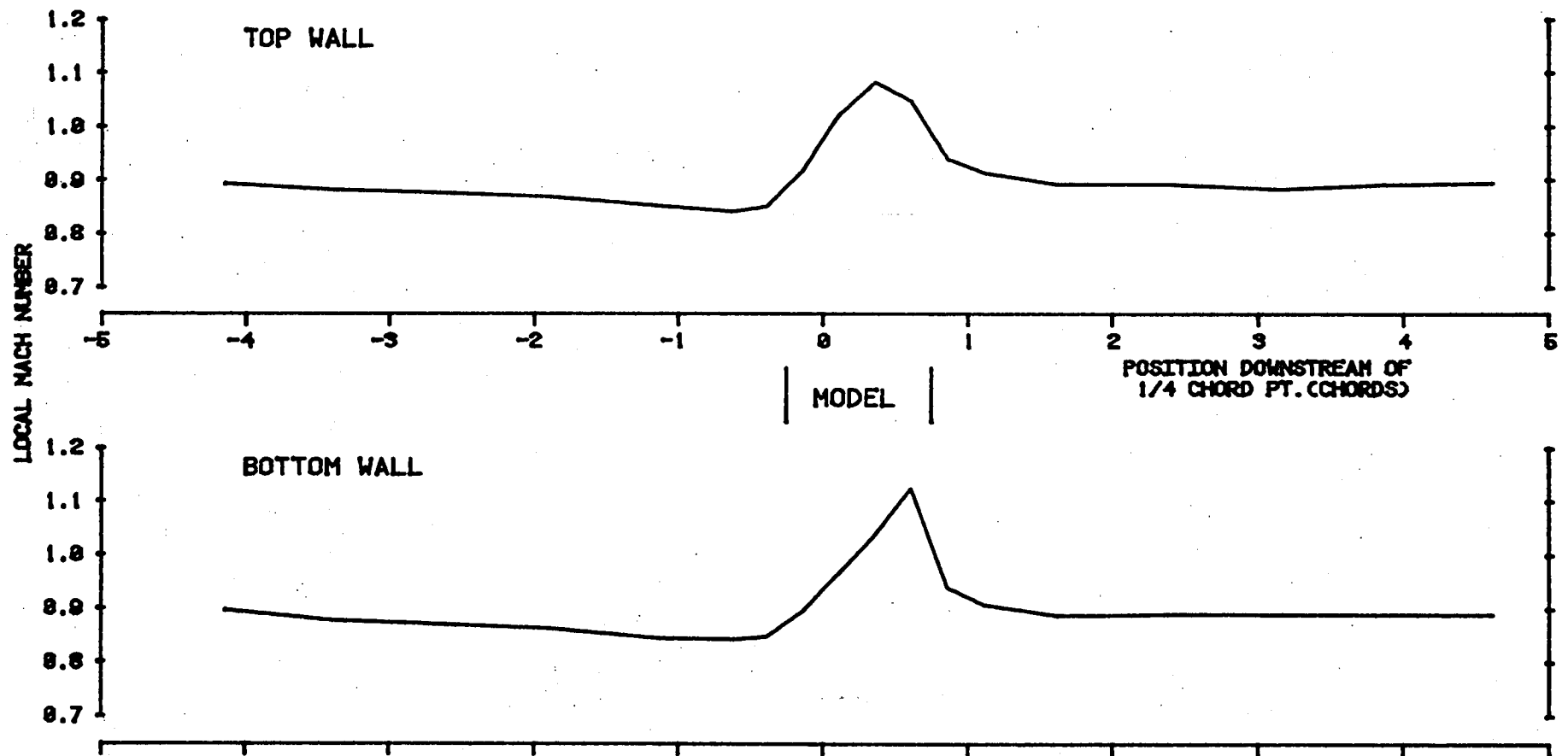


FIGURE 2.3

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
108 0.0 0.866

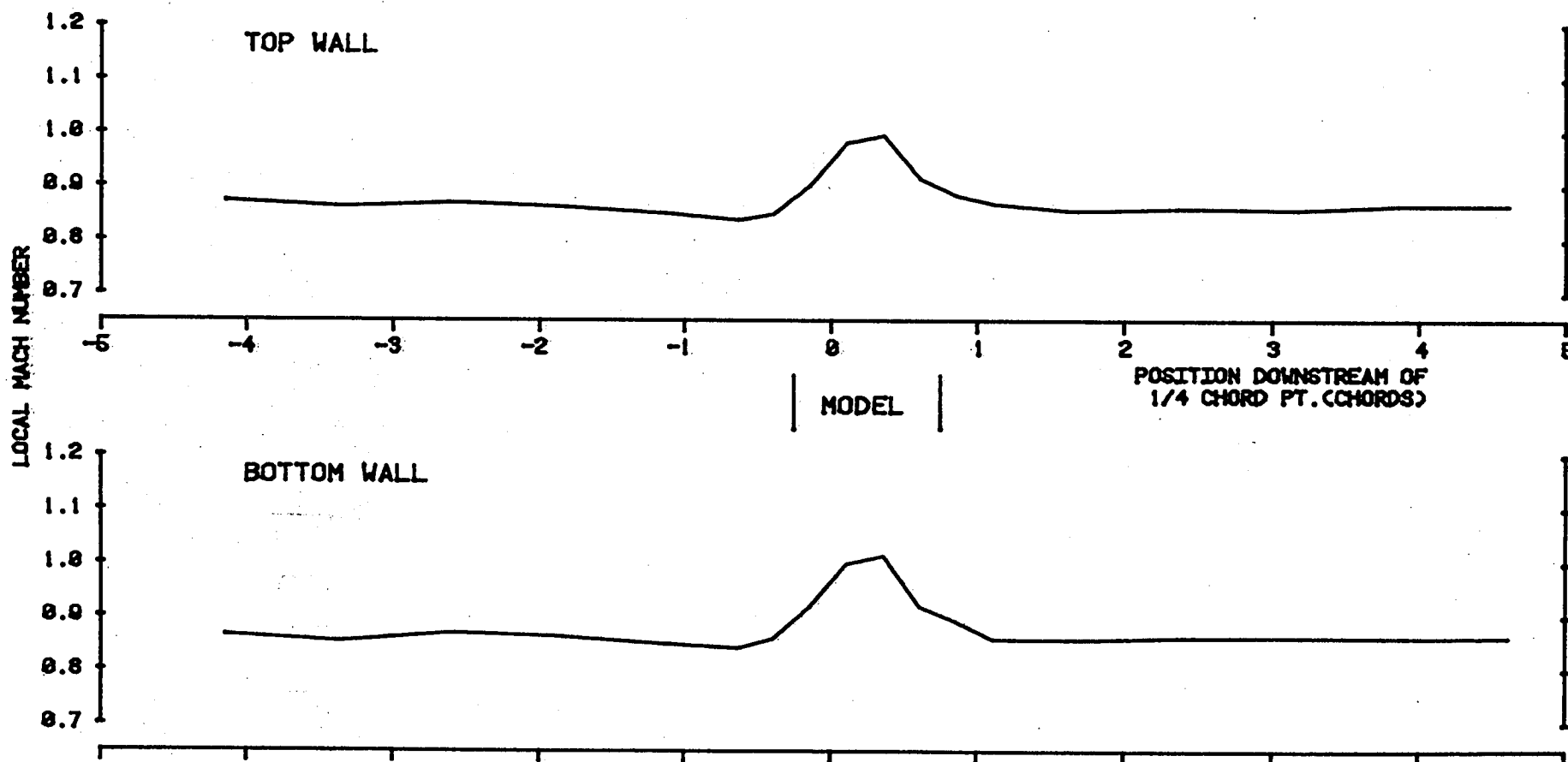


FIGURE 2.4

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
168 4.5 0.846

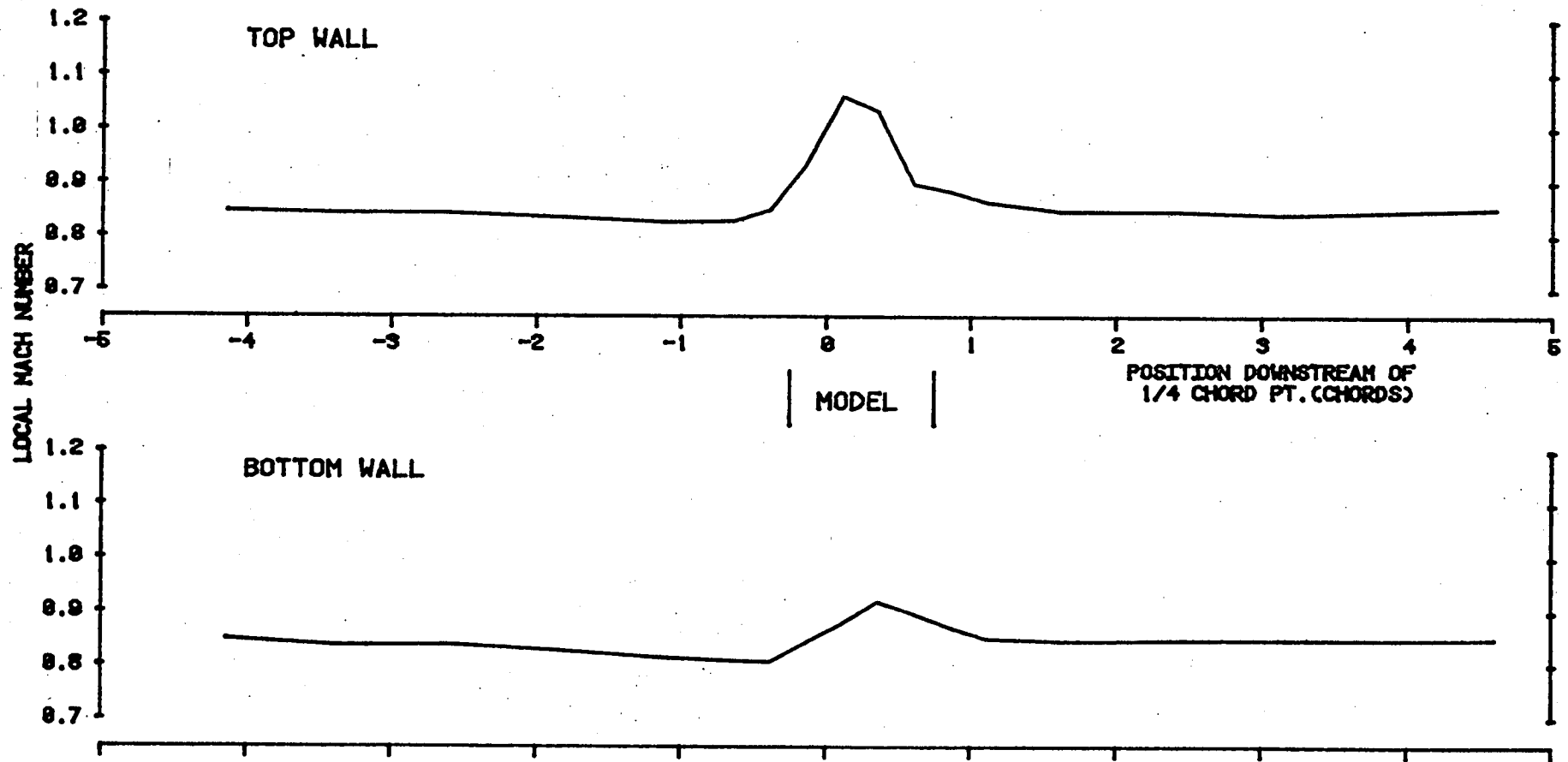


FIGURE 2.5

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
178 4.5 0.849

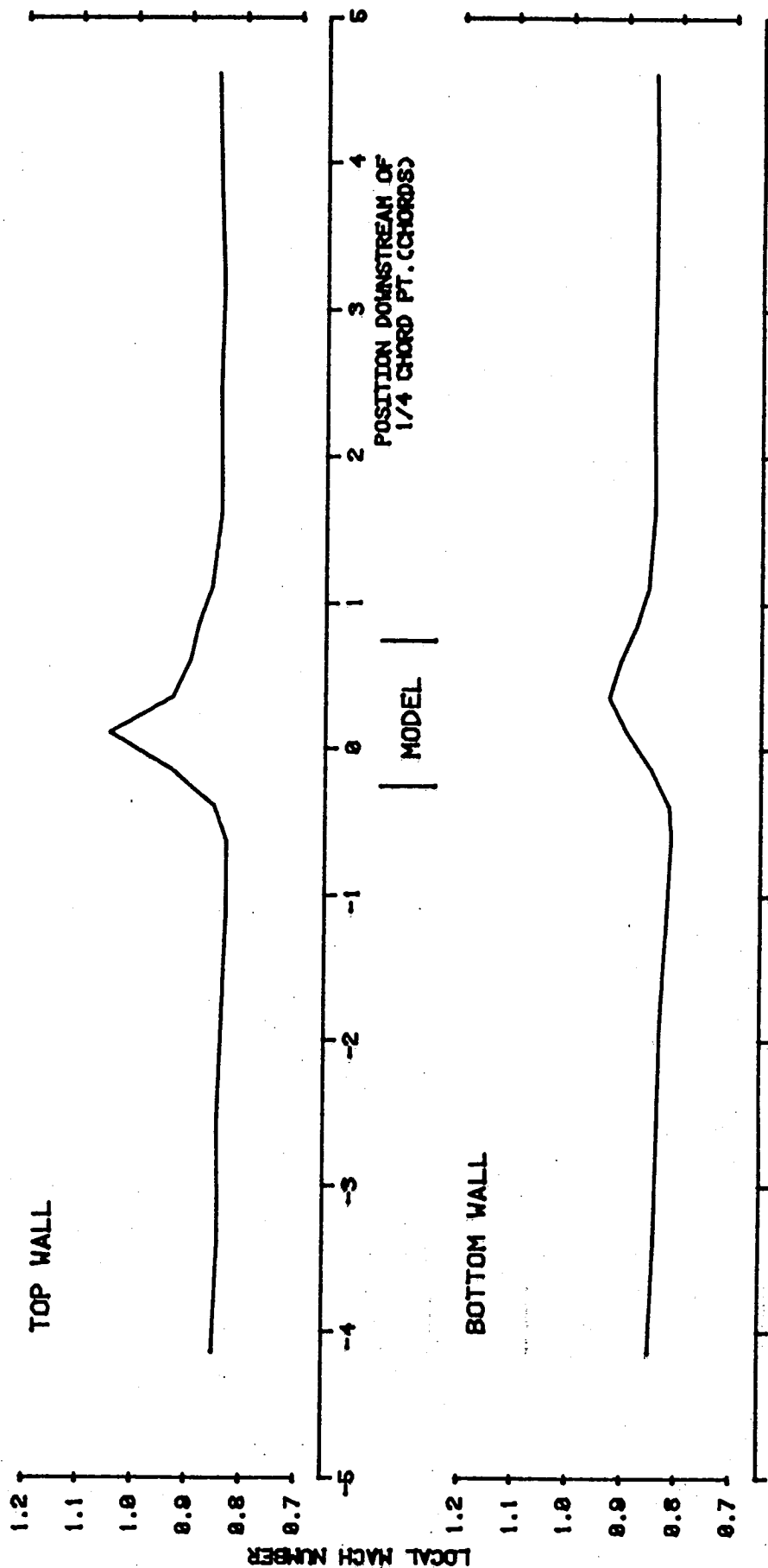


FIGURE 2.6

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
172 2.0 0.848

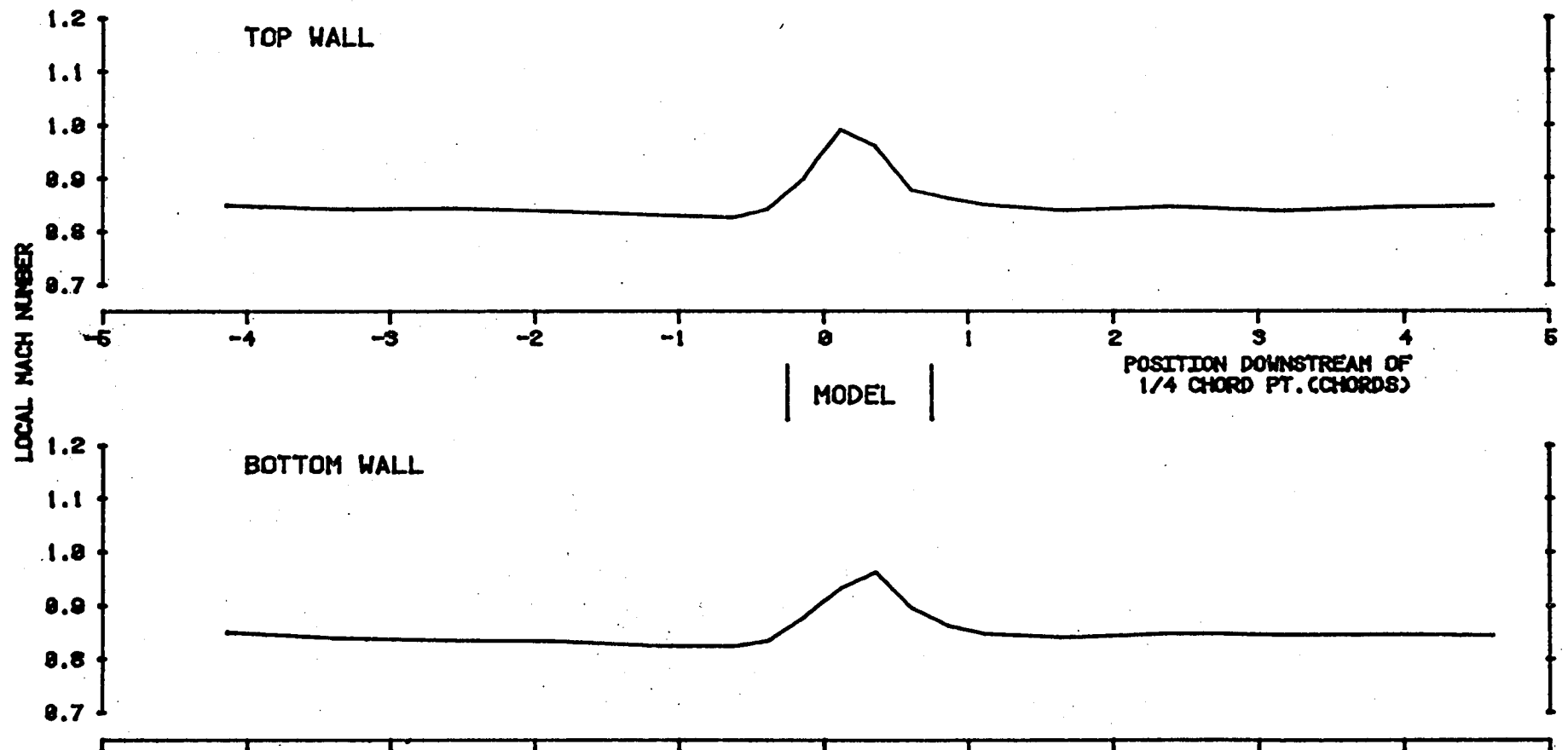


FIGURE 2.7

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
162 2.0 0.839

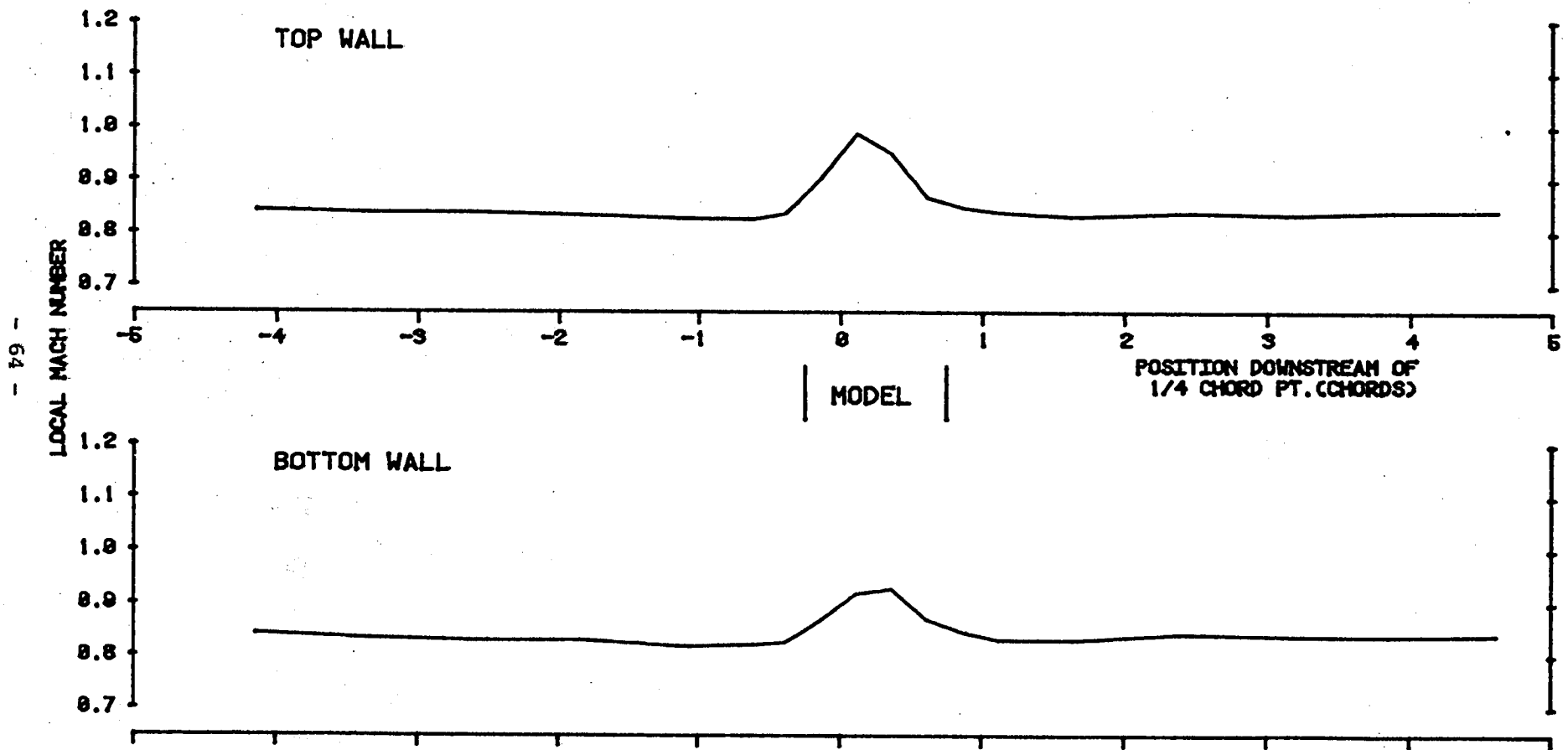


FIGURE 2.8

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
100 2.0 0.840

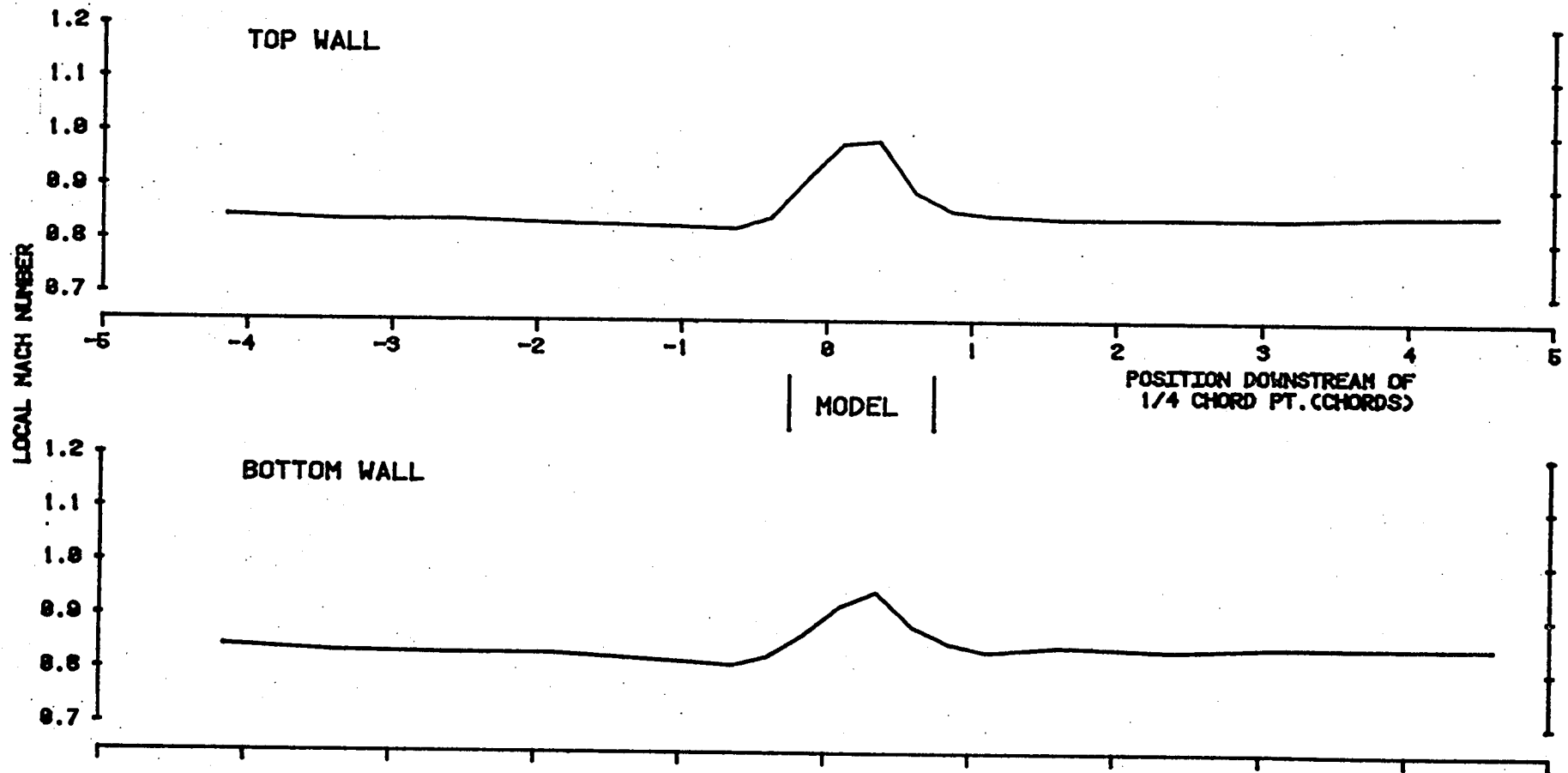


FIGURE 2.9

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO 136 ALPHA 0.0 MACH NO 0.854

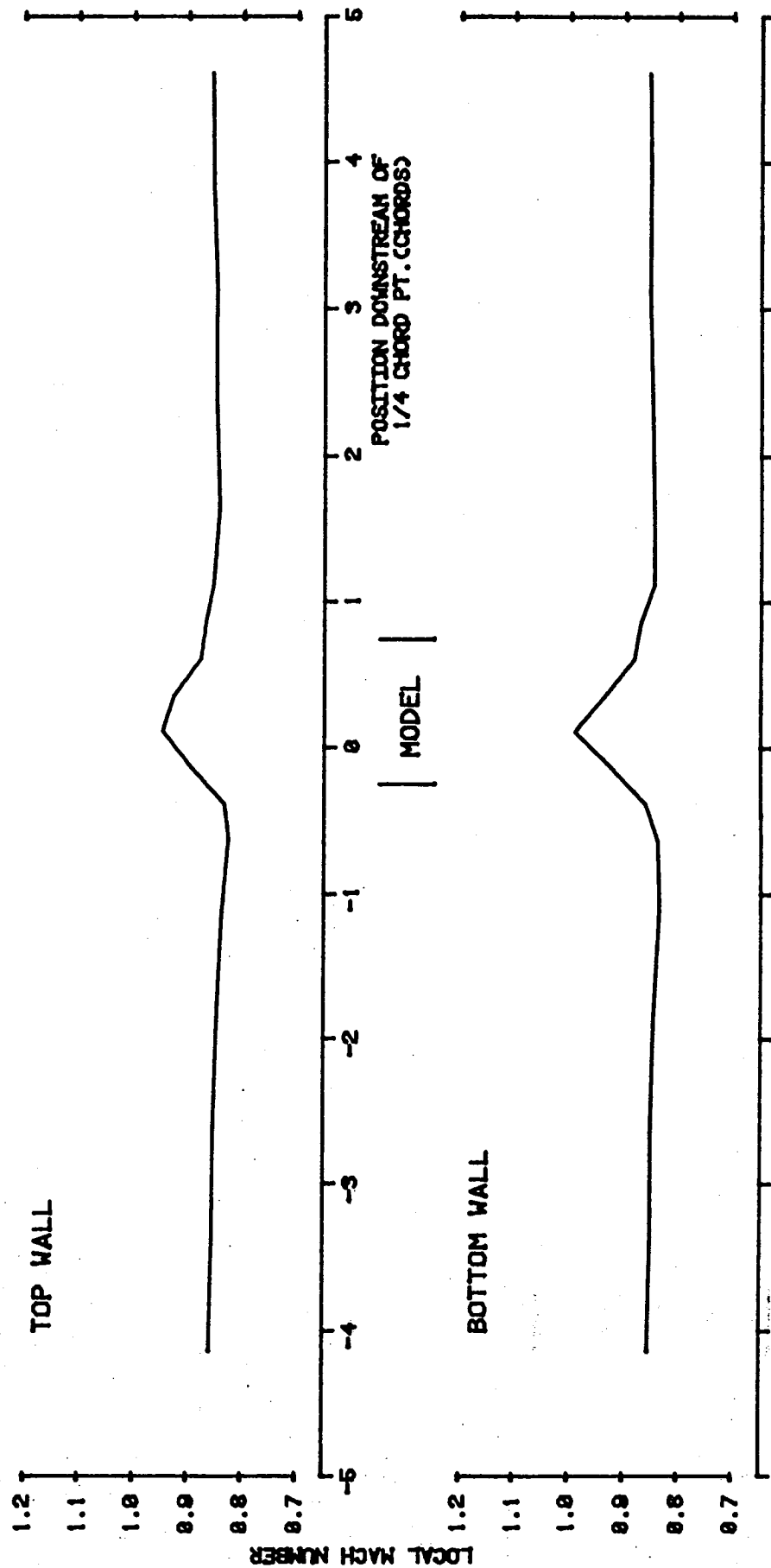


FIGURE 2.10

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
86 2.0 0.806

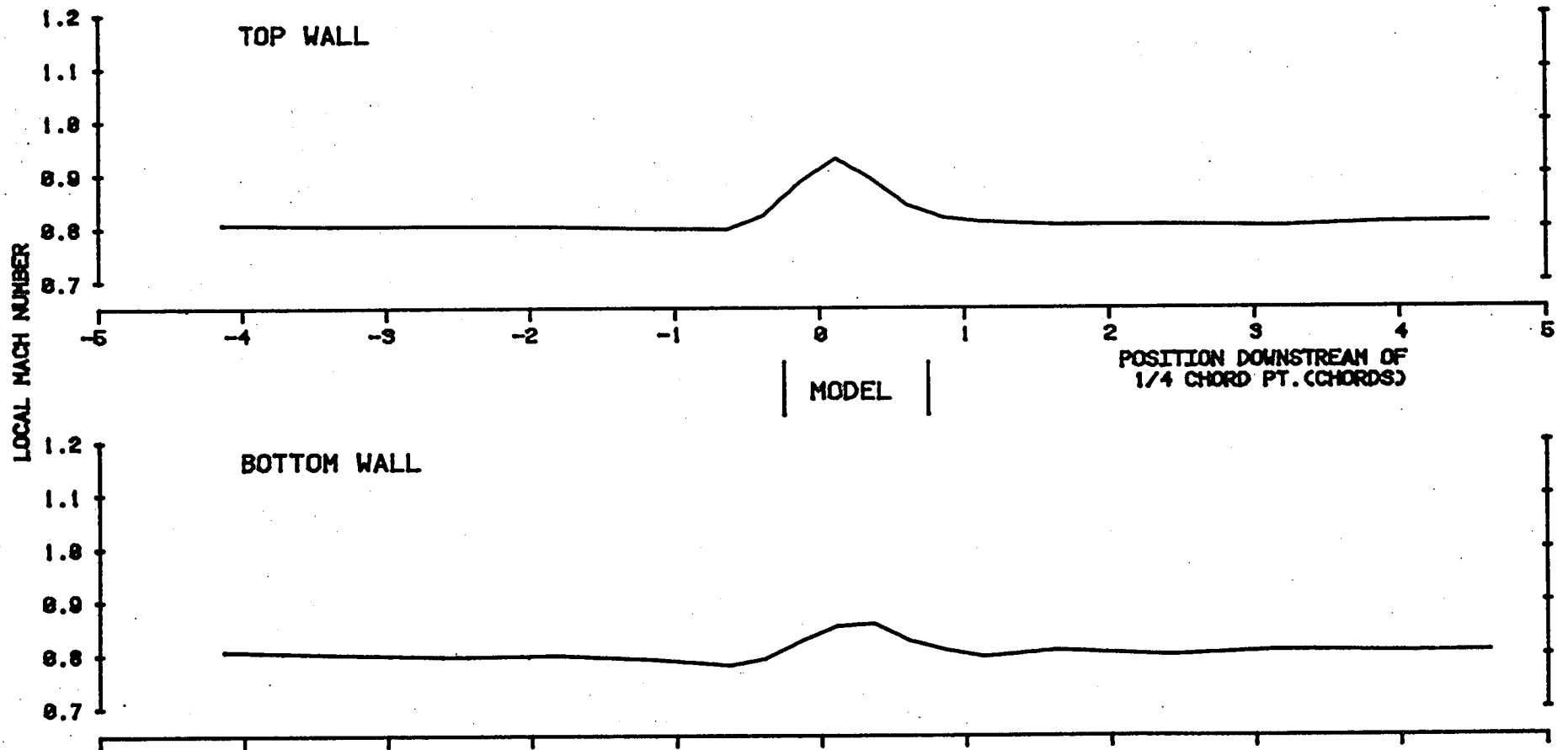


FIGURE 2.11

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
188 0.0 0.796

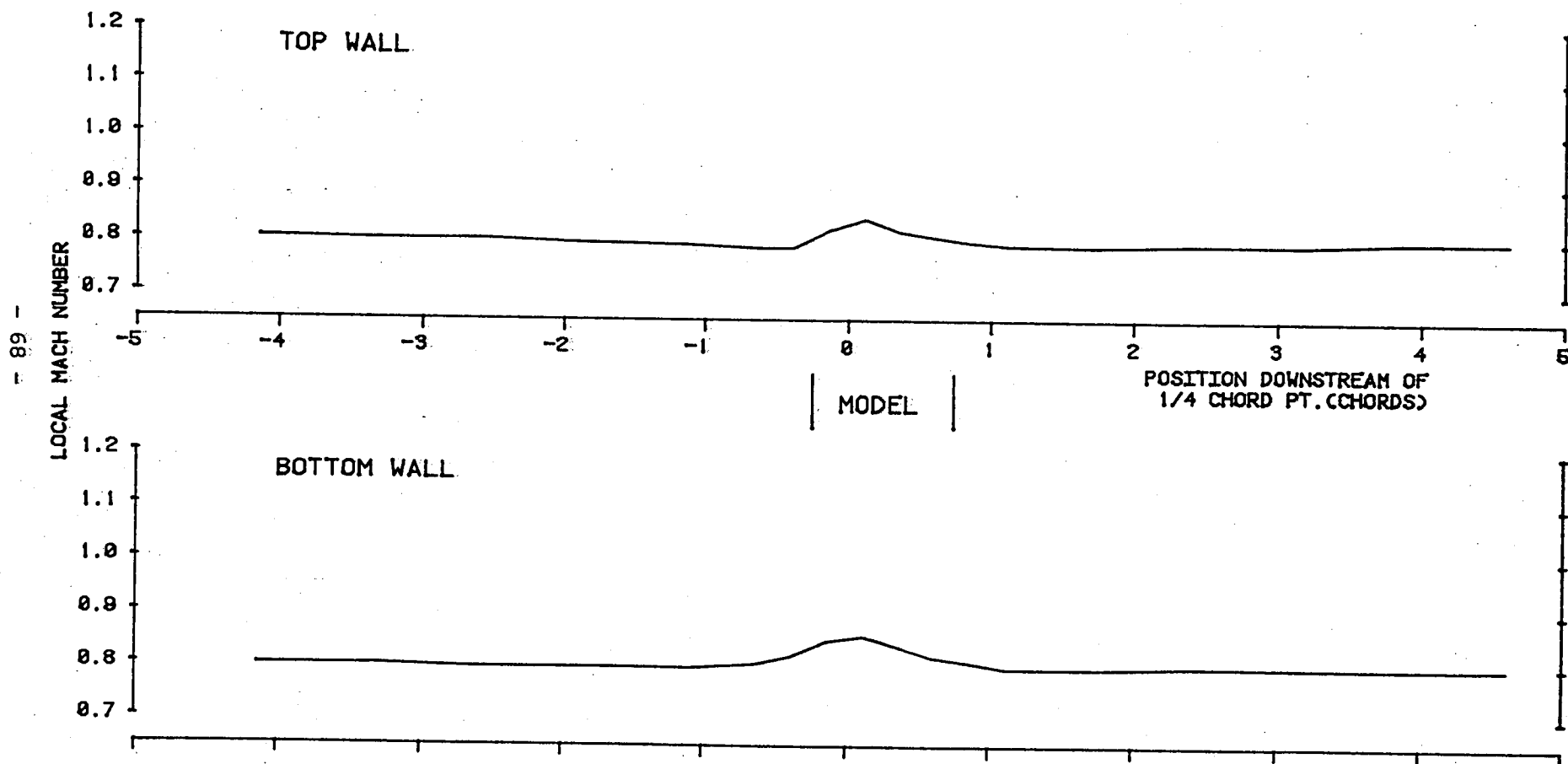


FIGURE 2.12

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO 105 ALPHA 0.0 MACH NO 0.753

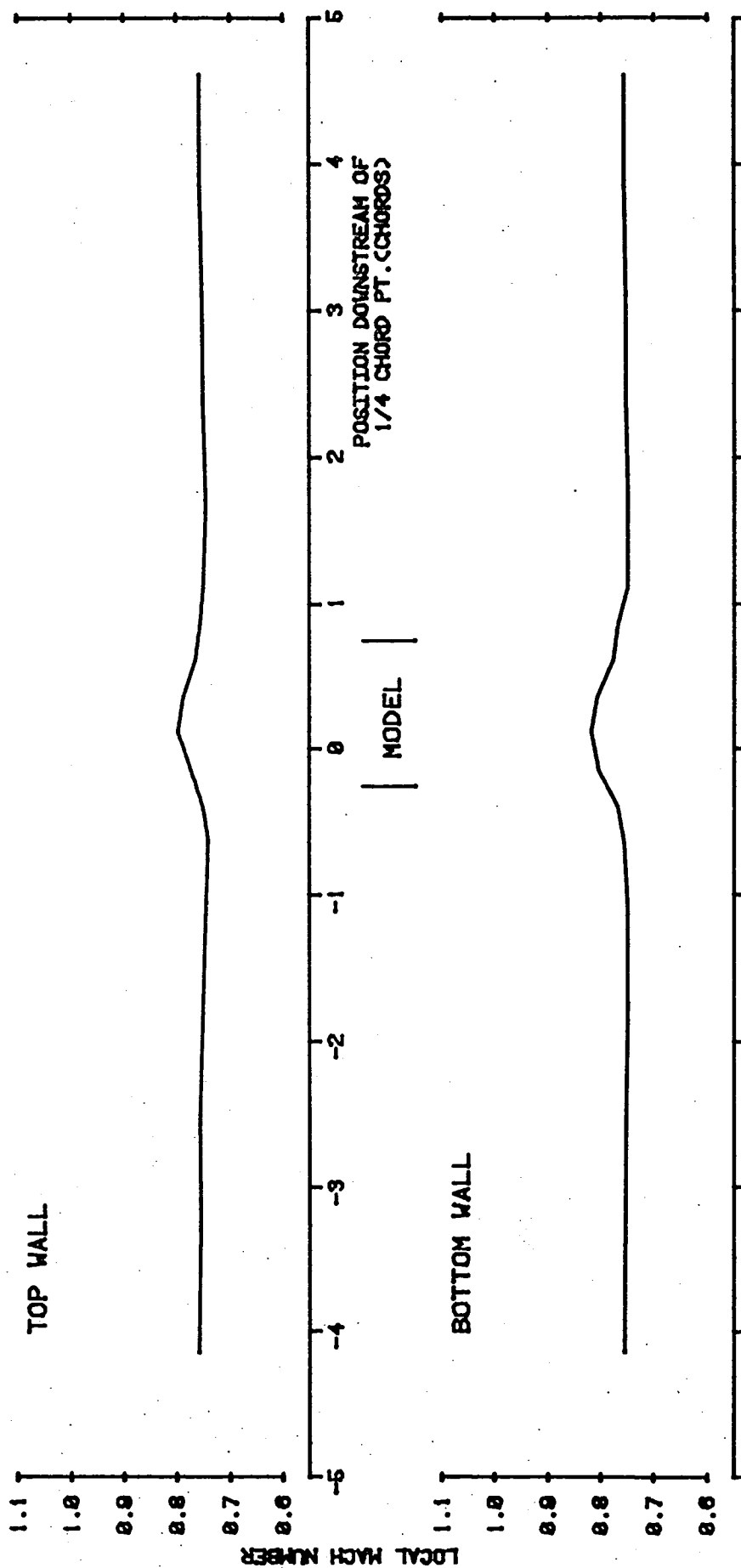


FIGURE 2.13

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
72 4.0 0.706

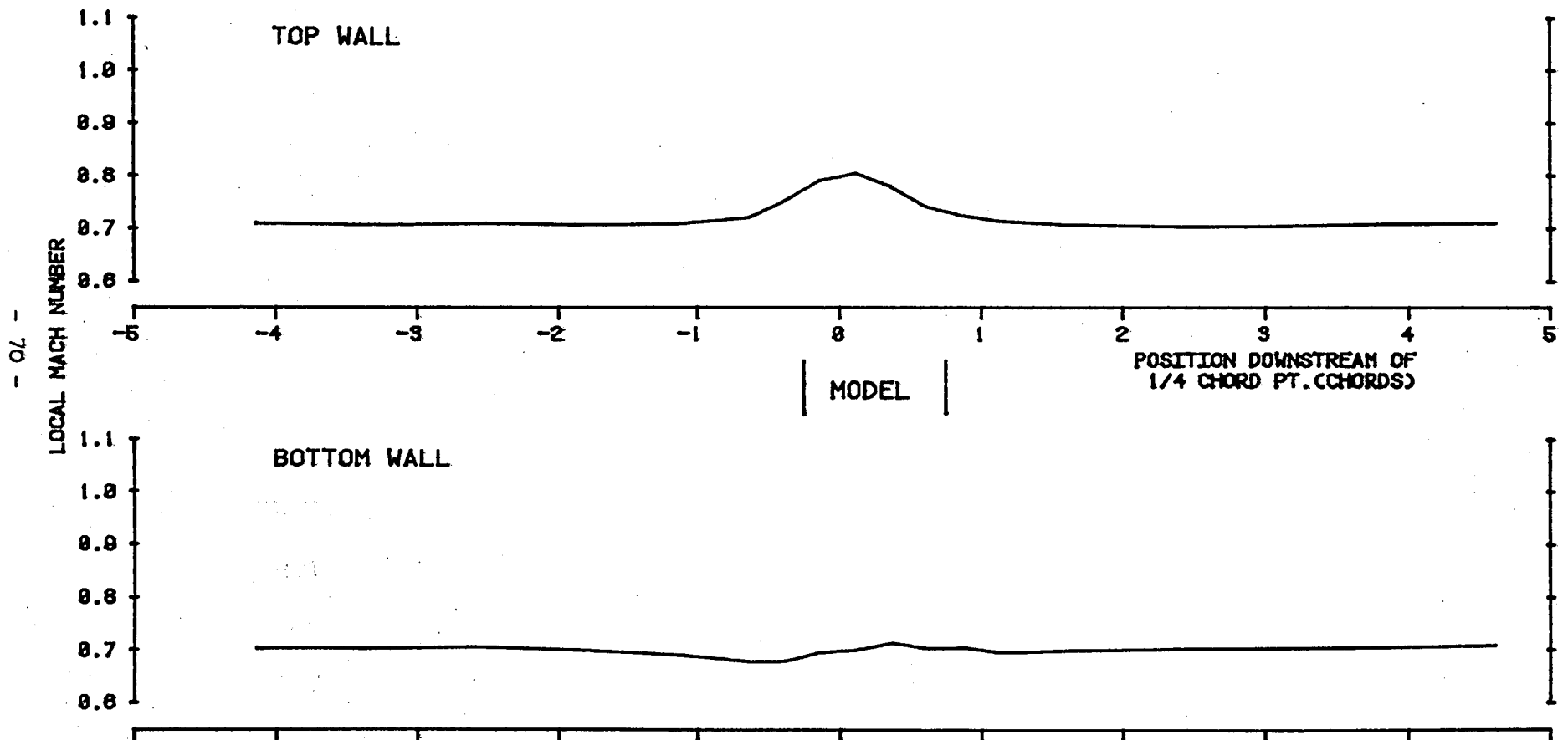


FIGURE 2.14

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
63 4.0 0.706

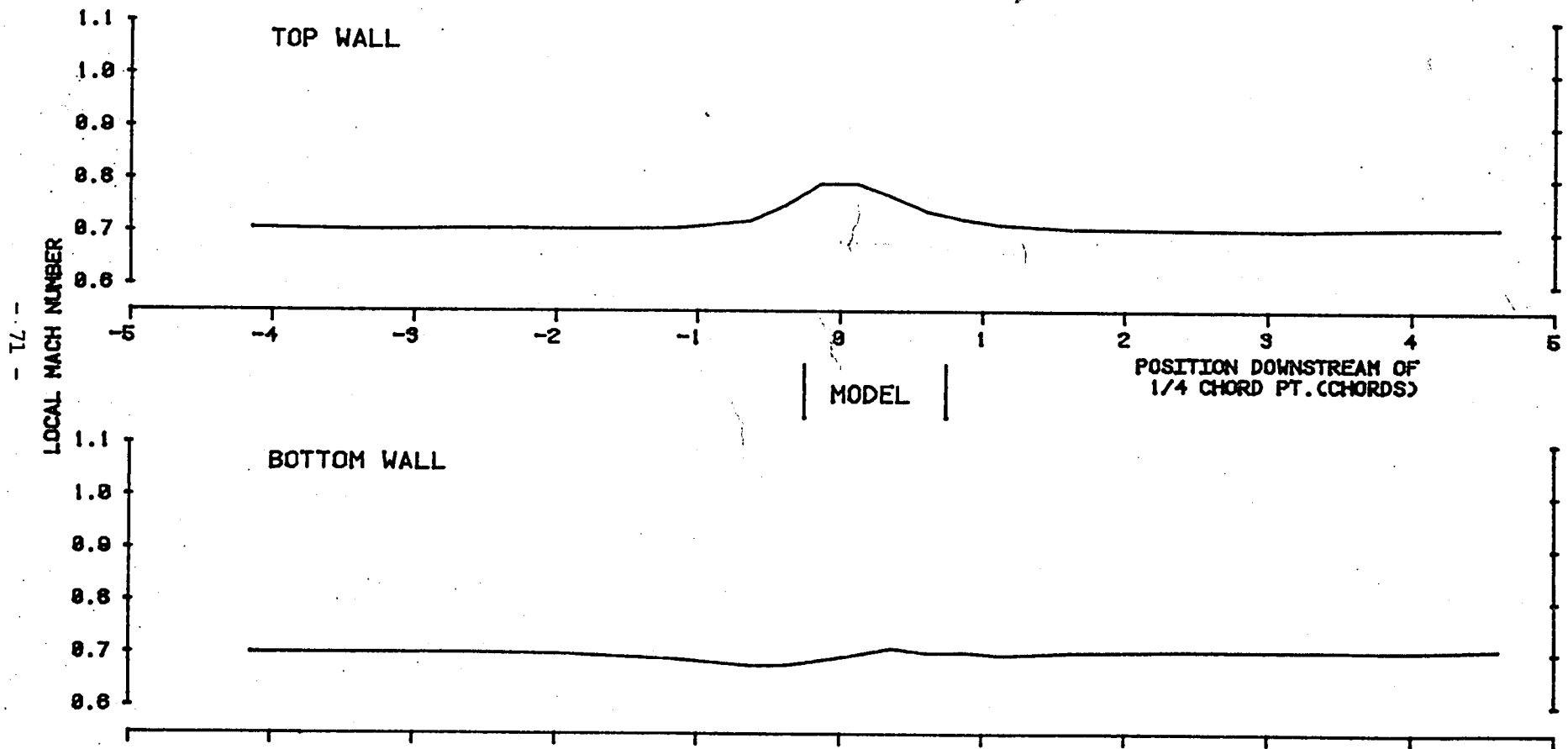


FIGURE 2.15

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO 69 ALPHA 3.0 MACH NO 0.701

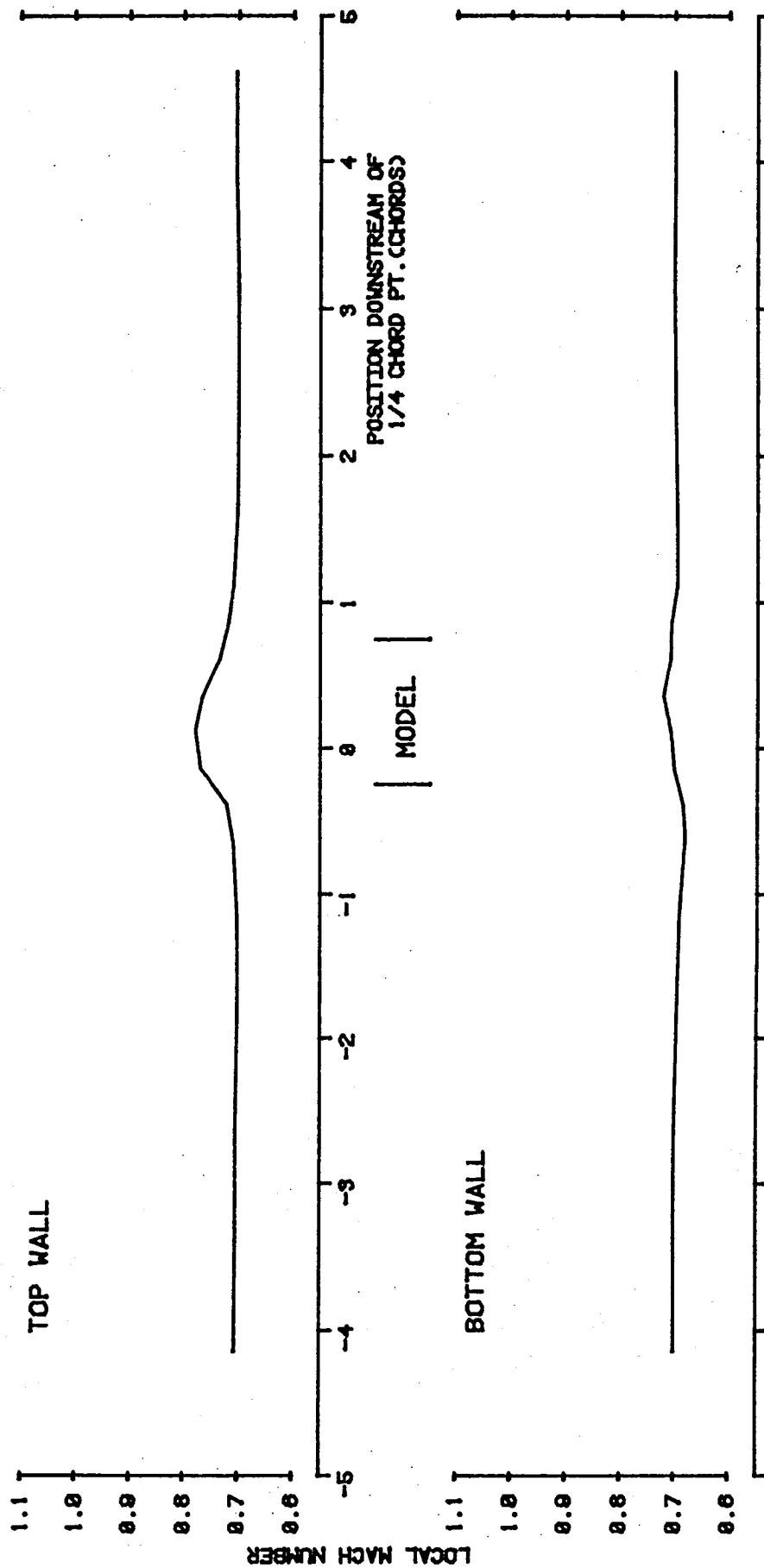


FIGURE 2.16

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
65 2.0 0.703

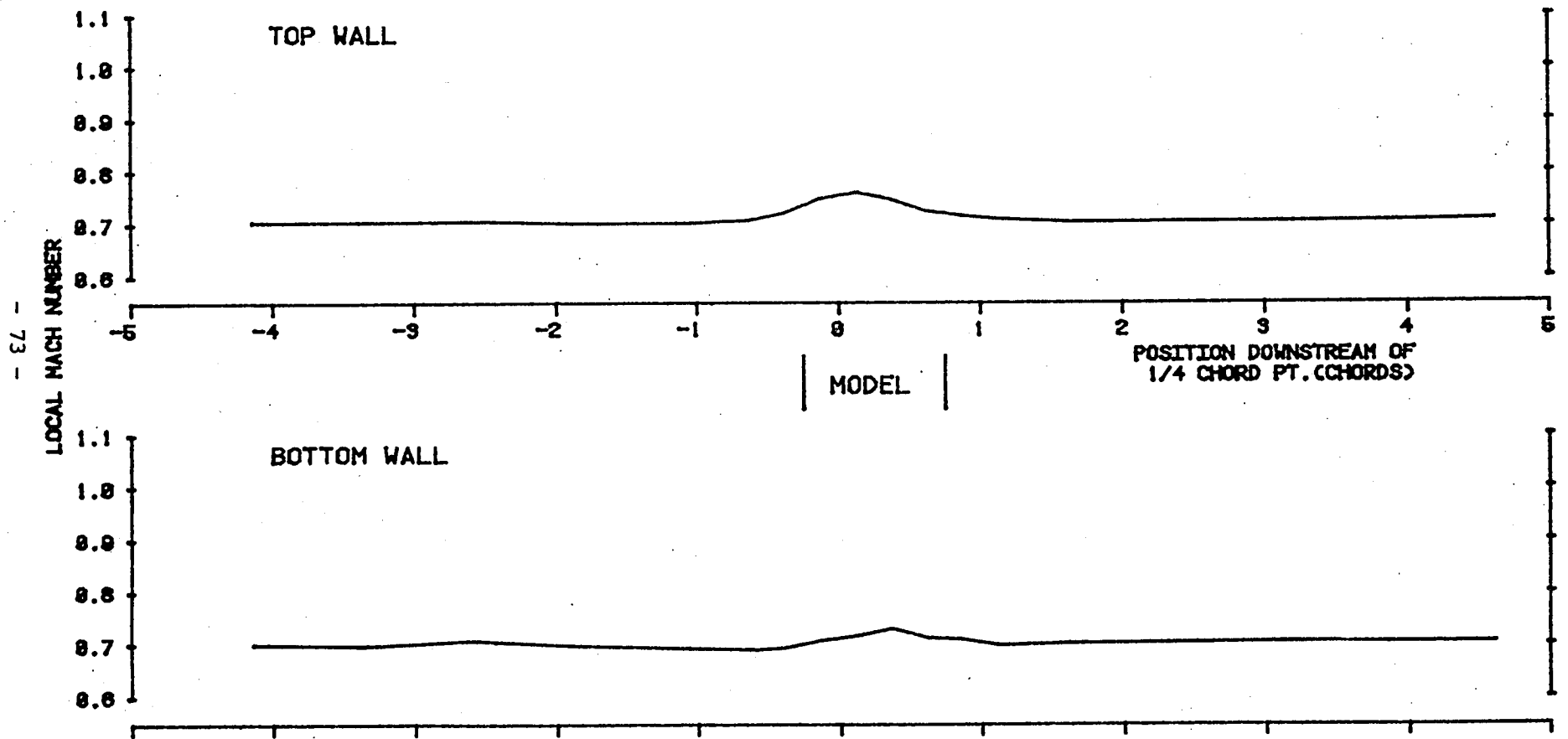


FIGURE 2.17

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
83 2.0 0.712

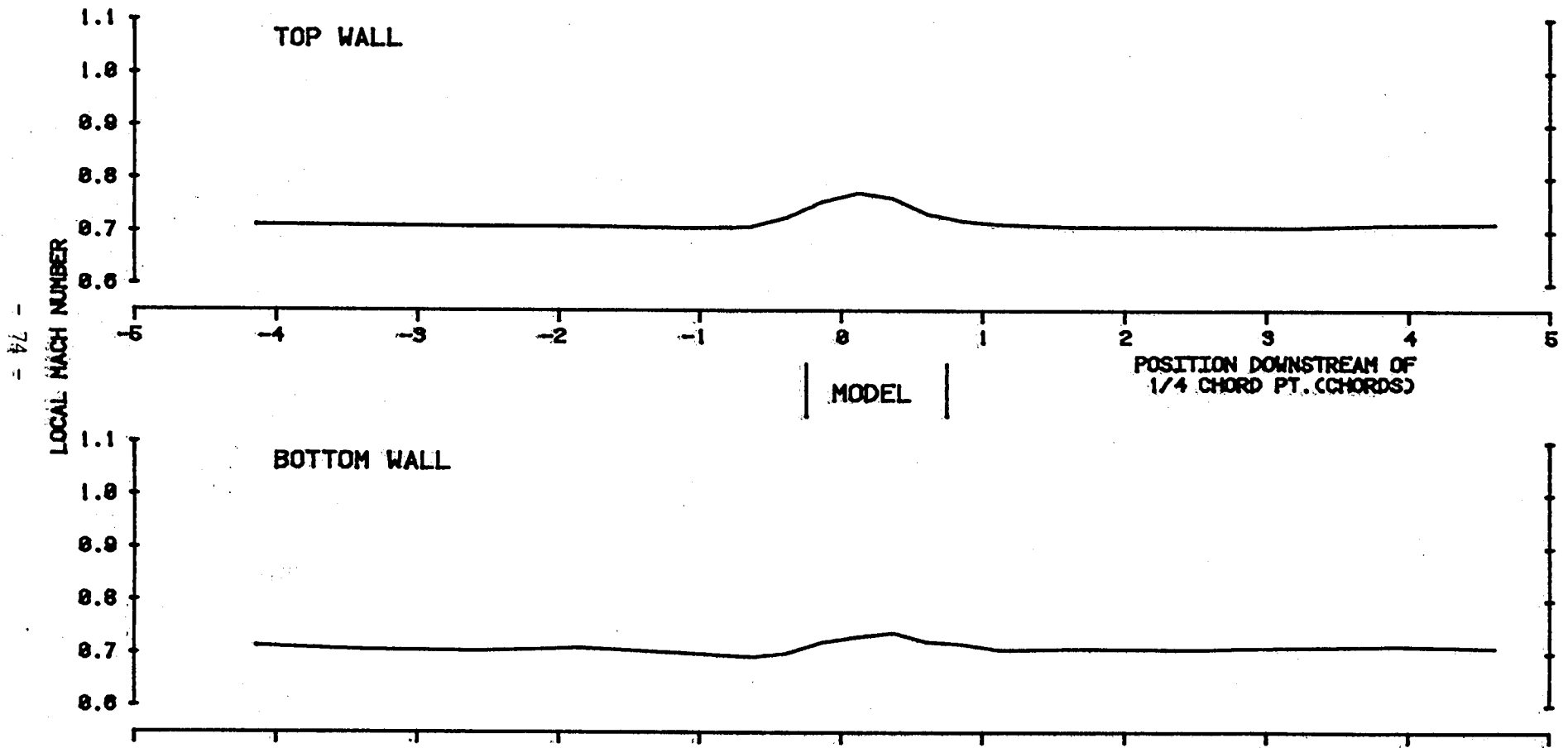


FIGURE 2.18

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO 122 ALPHA 0.0 MACH NO 0.688

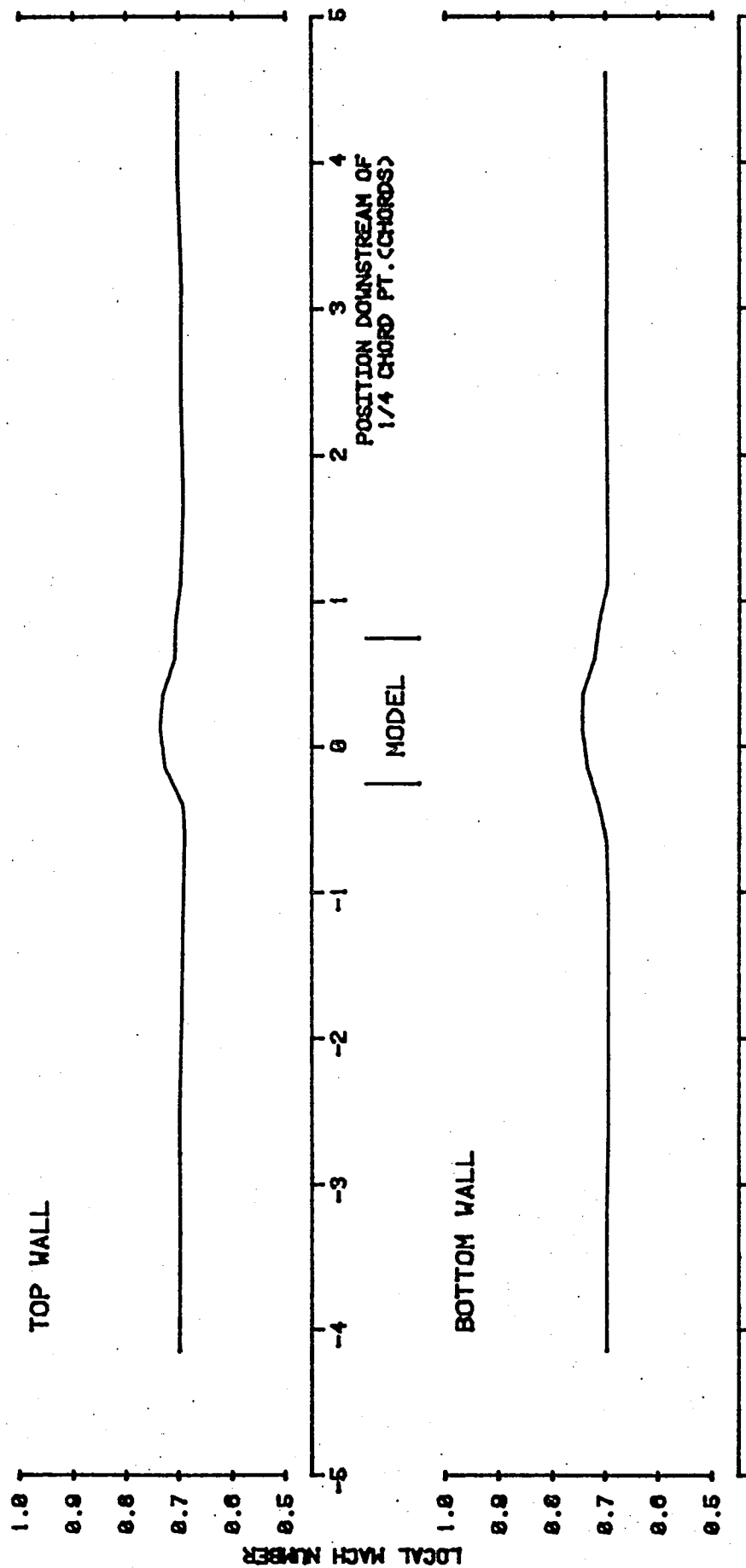


FIGURE 2.19

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
115 6.0 0.506

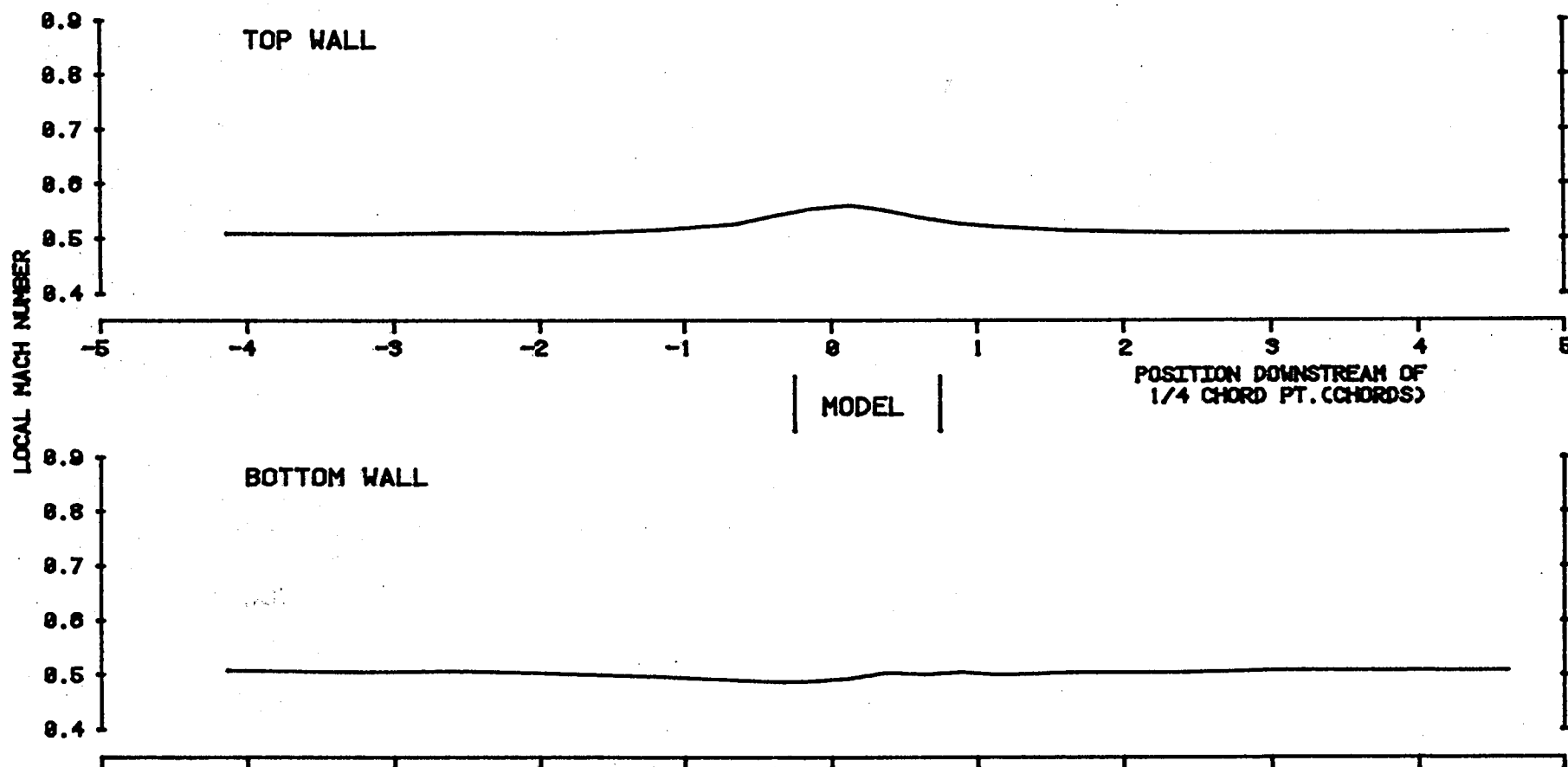


FIGURE 2.20

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
112 4.0 0.507

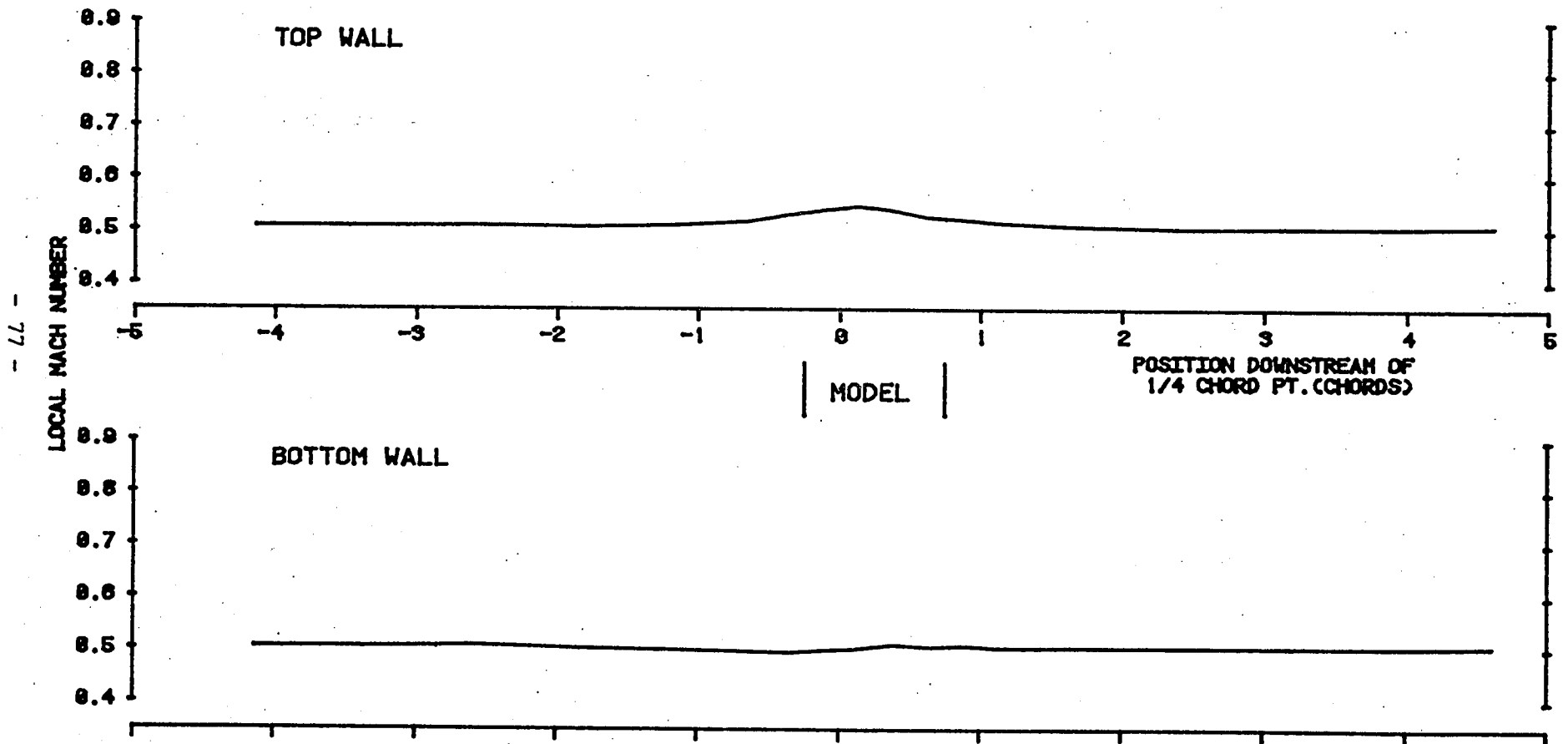


FIGURE 2.21

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO 91 ALPHA 2.0 MACH NO 0.508

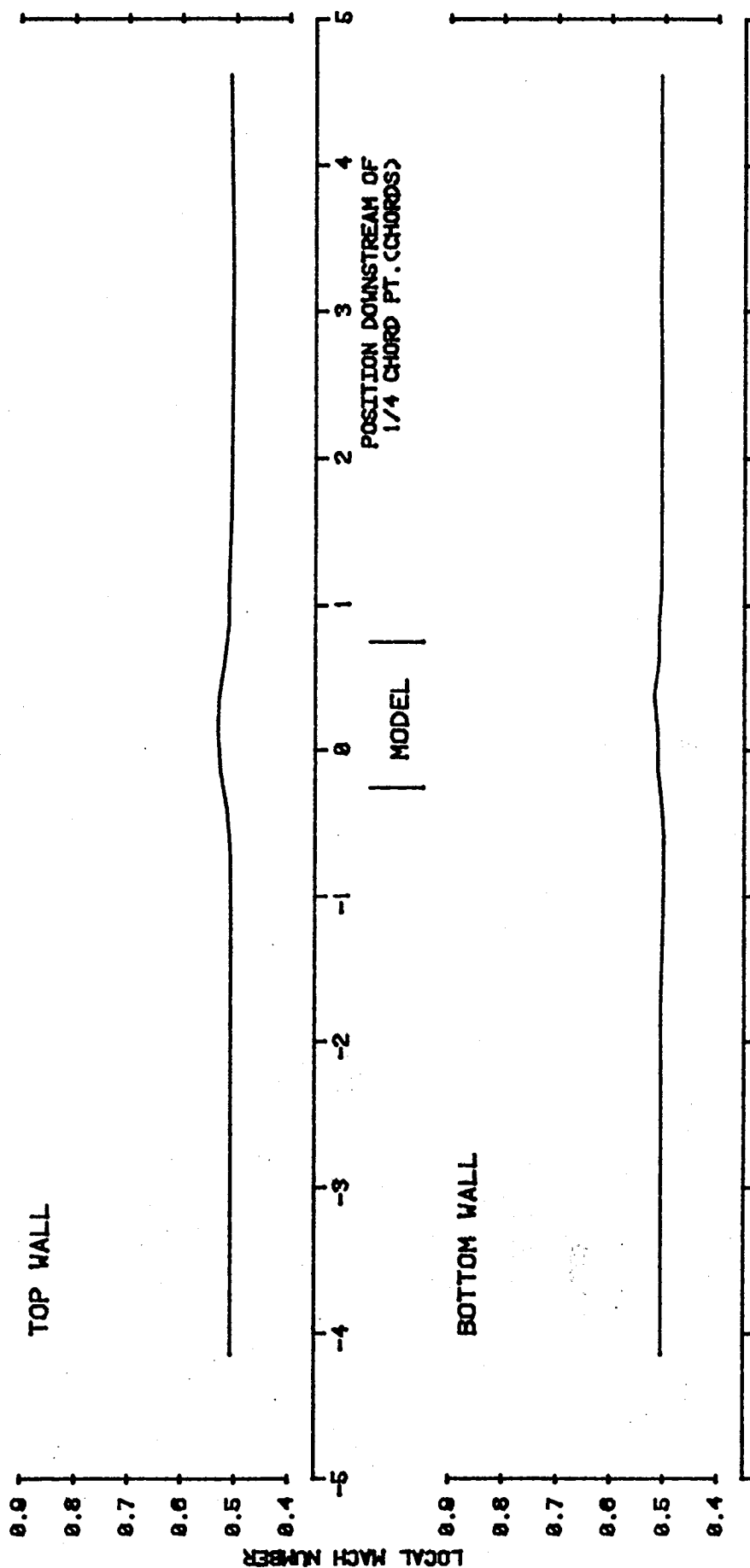


FIGURE 2.22

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
109 2.0 0.594

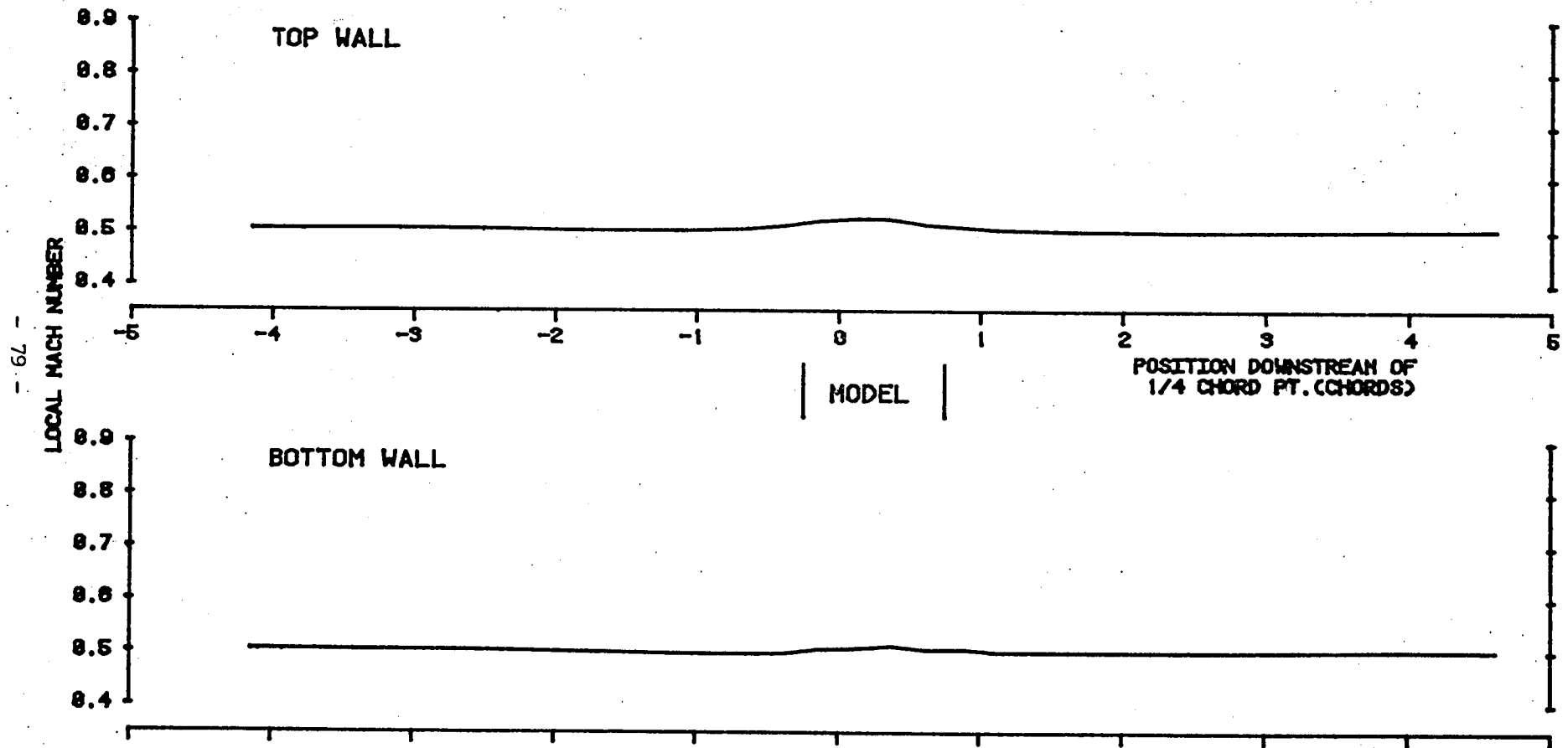


FIGURE 2.23

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
105 0.0 0.506

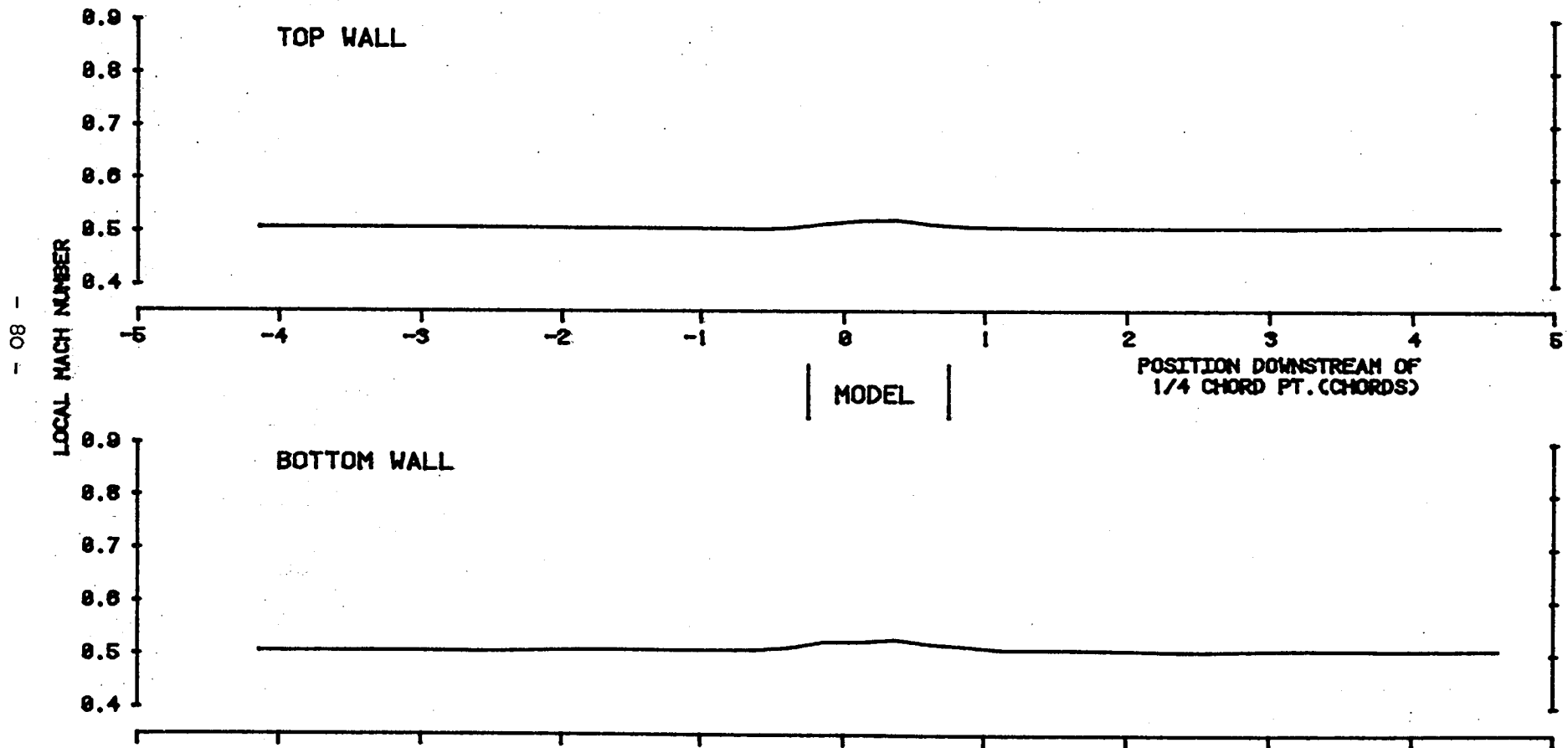


FIGURE 2.24

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
89 2.0 0.306

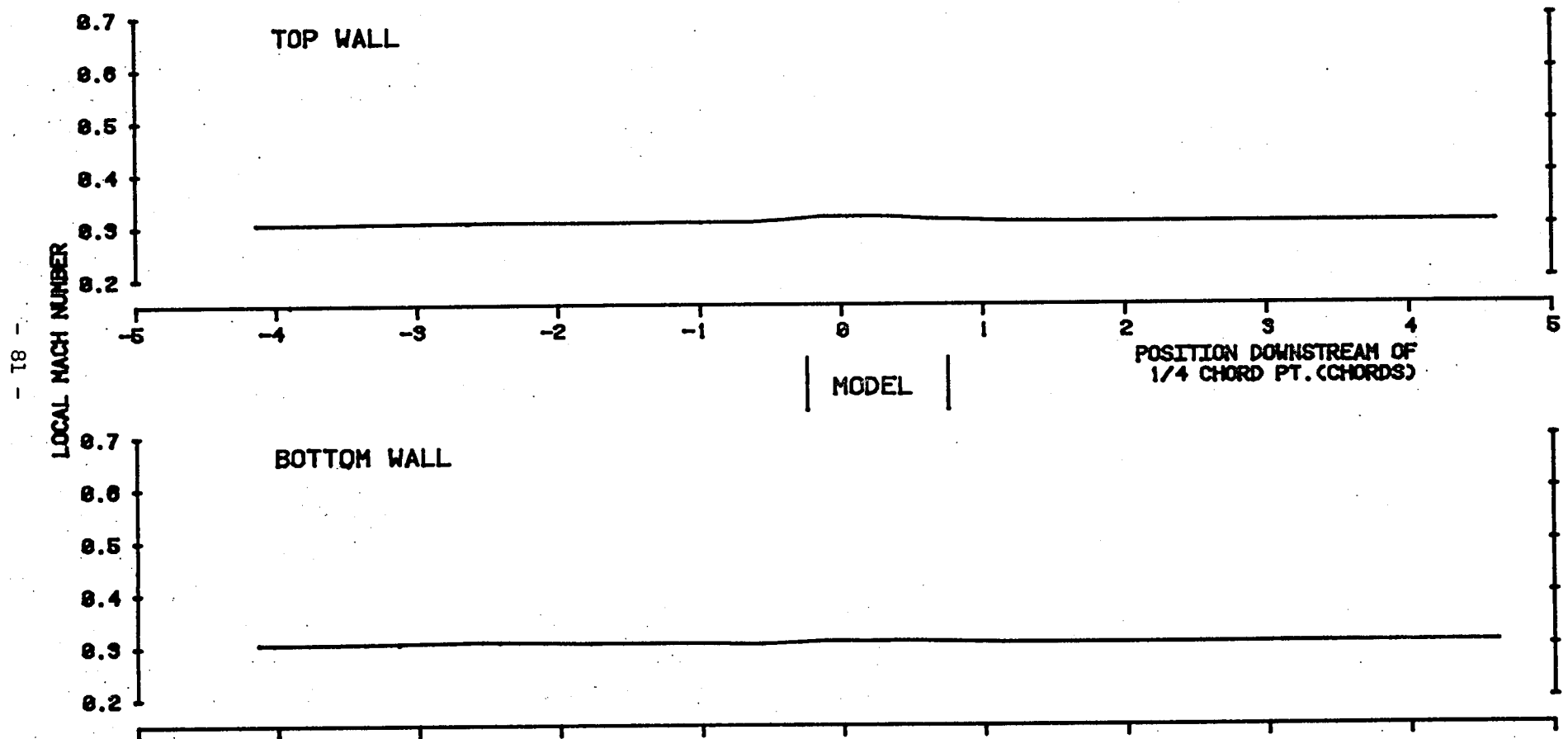


FIGURE 2.25

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
224 4.0 0.884

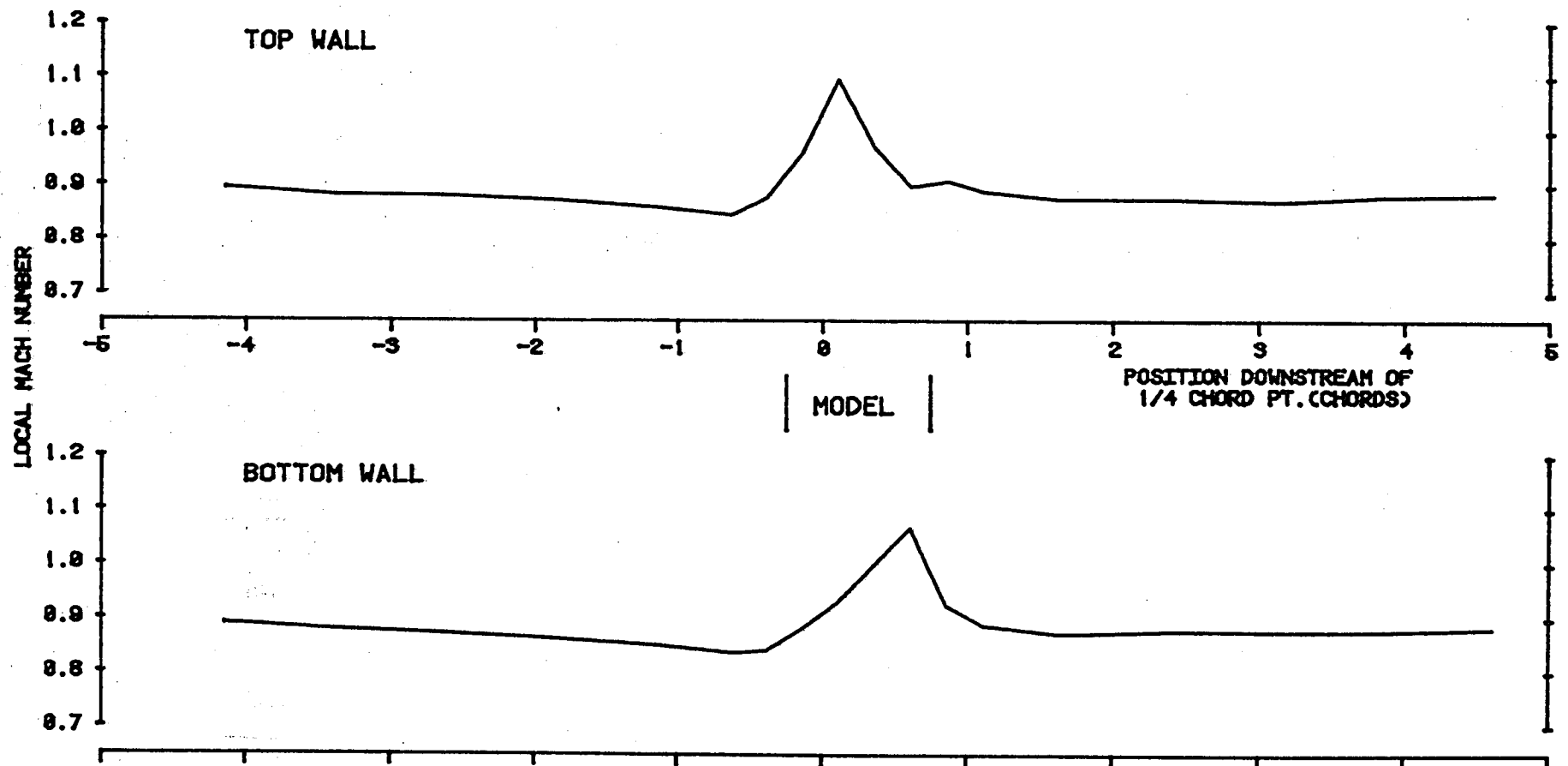


FIGURE 2.26

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO 208 ALPHA 0.0 MACH NO 0.889

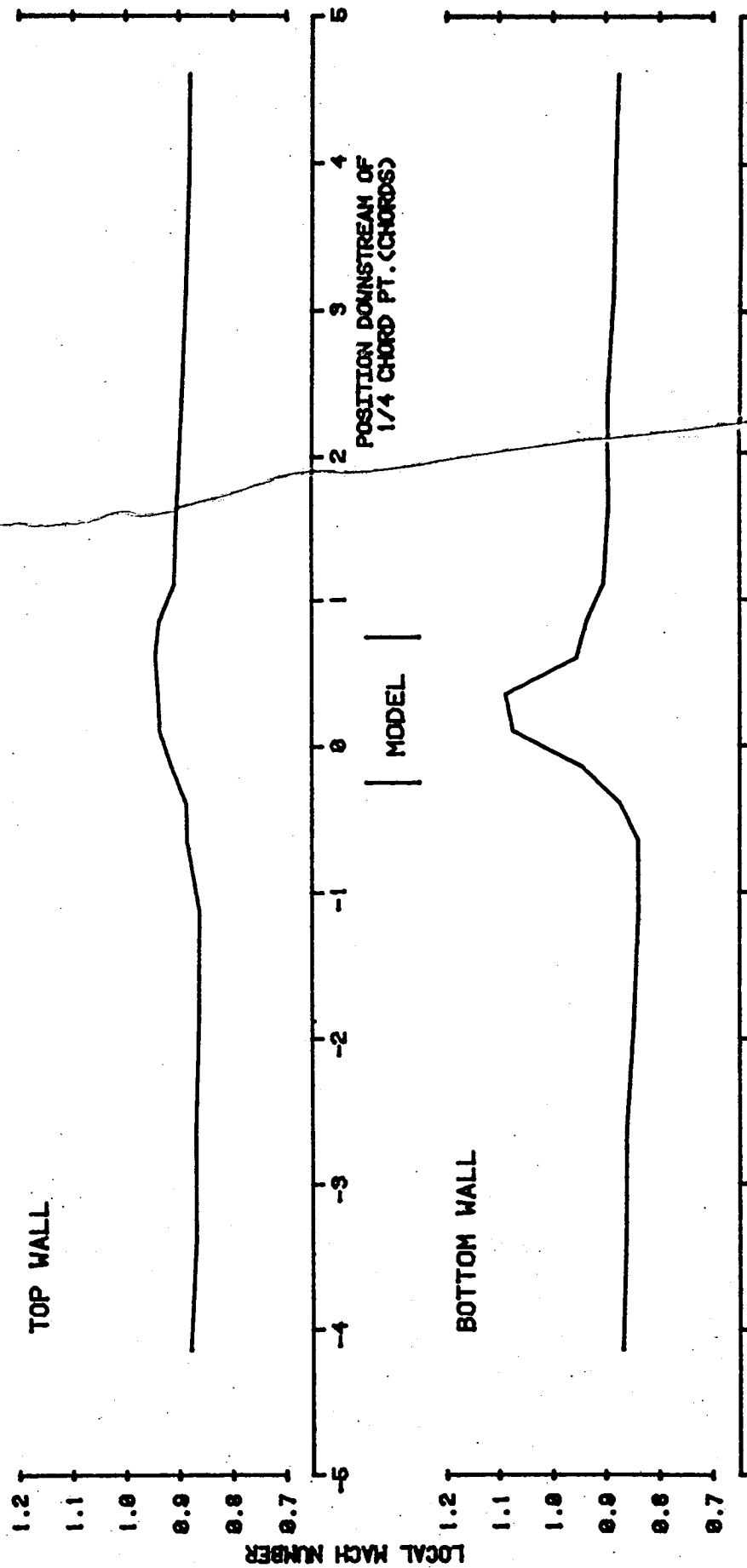


FIGURE 2.27

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
215 0.0 0.841

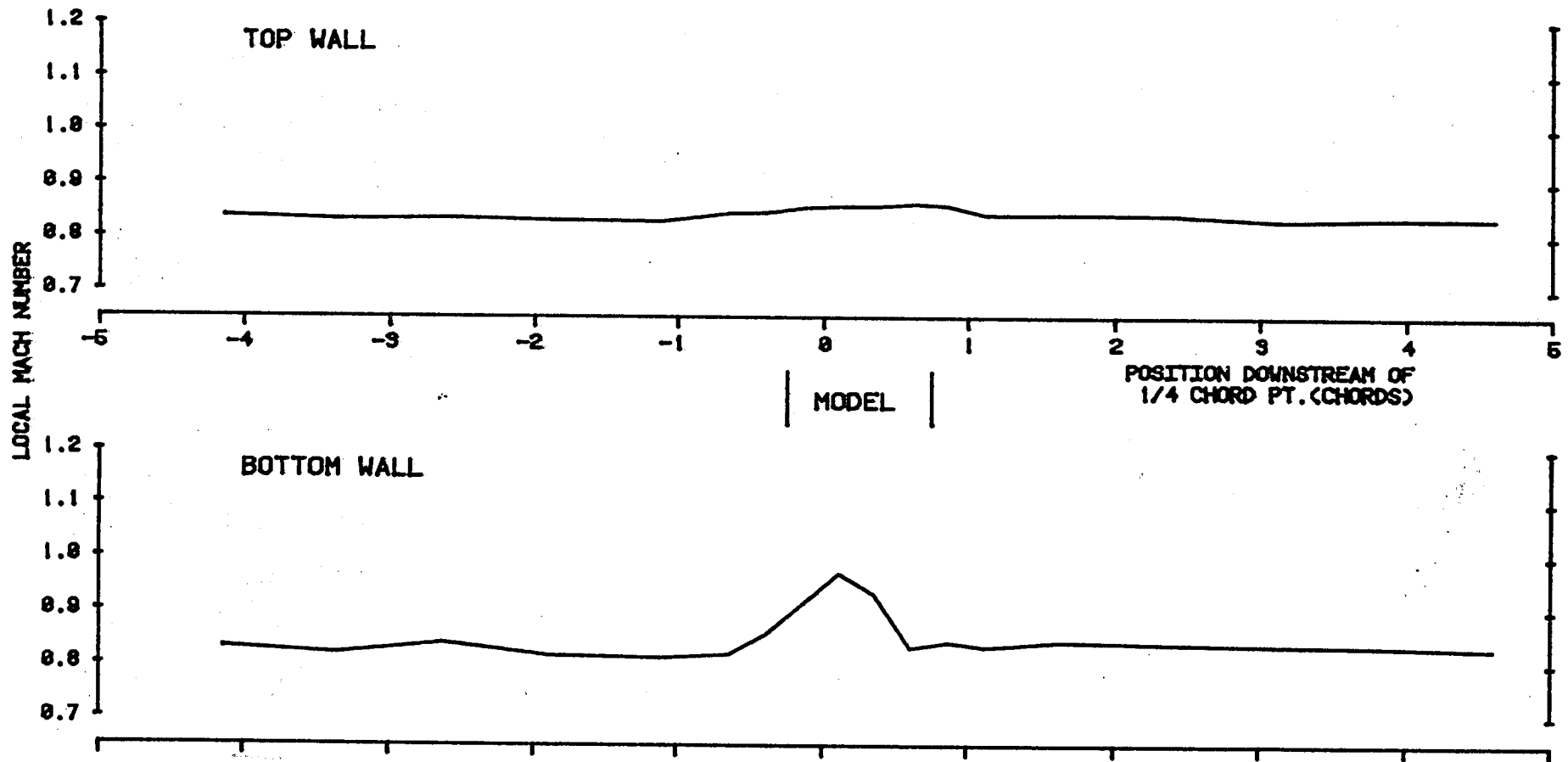


FIGURE 2.28

TSWT MACH NO. DISTRIBUTION  
ALONG FLEXIBLE WALLS

RUN NO ALPHA MACH NO  
219 0.0 0.708

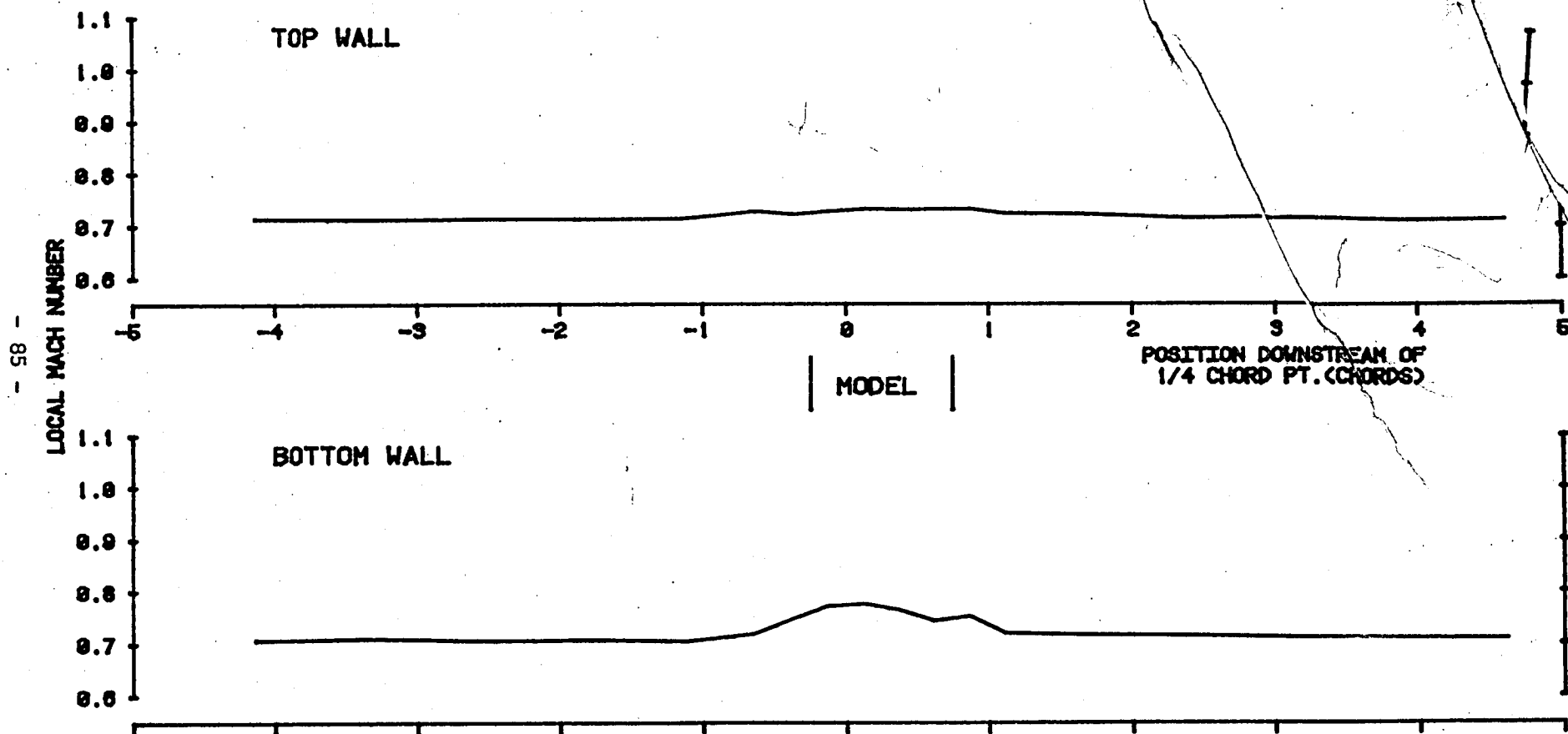


FIGURE 2.29

TSWT MACH NO. DISTRIBUTION  
ALONG STRAIGHT FLEXIBLE WALLS

RUN NO MACH NO  
195 0.899  
30 0.303

+ TOP WALL

△ BOTTOM WALL

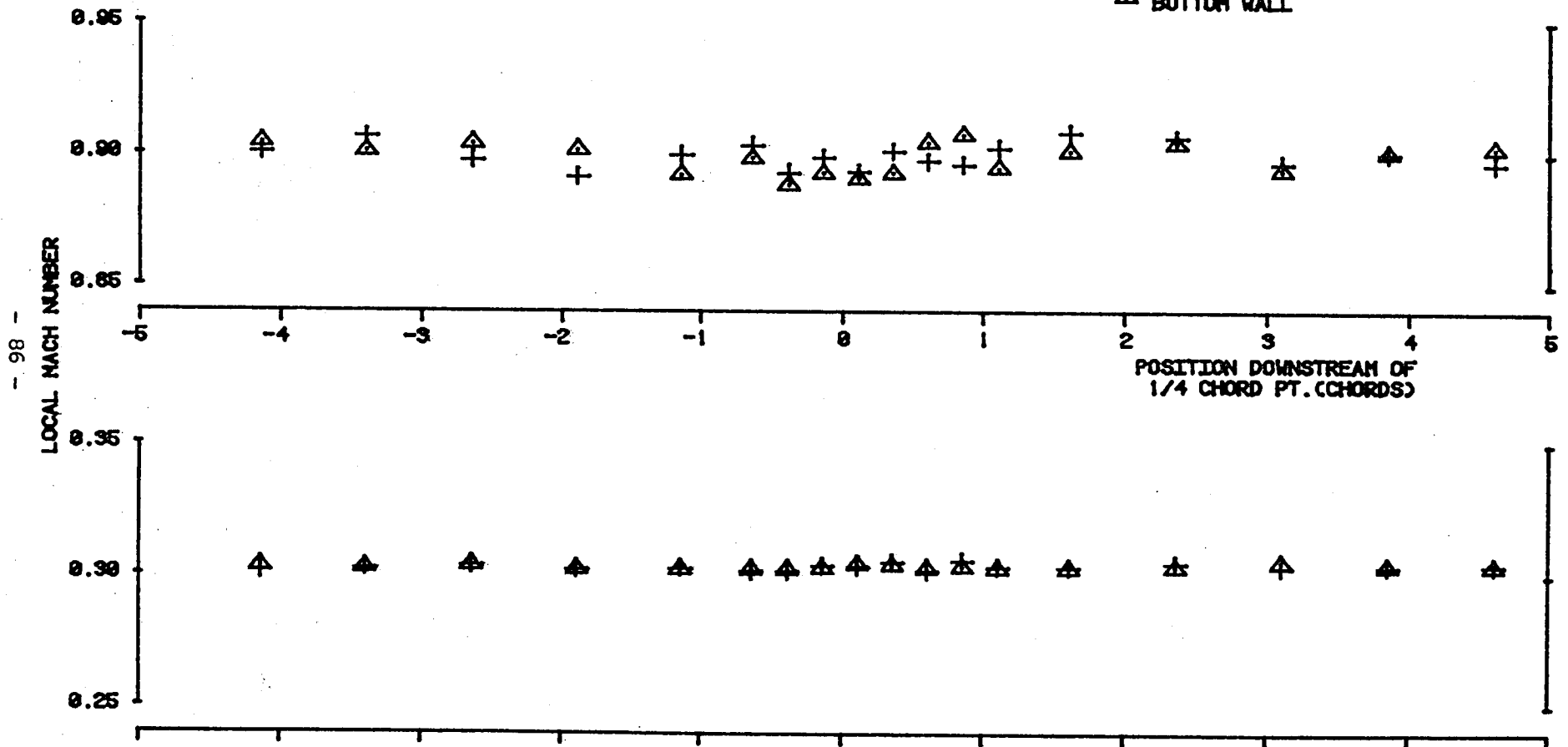


FIGURE 3

FLEXIBLE WALL EFFECTIVE AERODYNAMIC

CONTOURS RELATIVE TO STRAIGHT WALL

CONTOURS

FIGURE 3.1

TSWT WALL DELTA\* CONTOURS

RUN NO 184 ALPHA 4.0 MACH NO 0.886

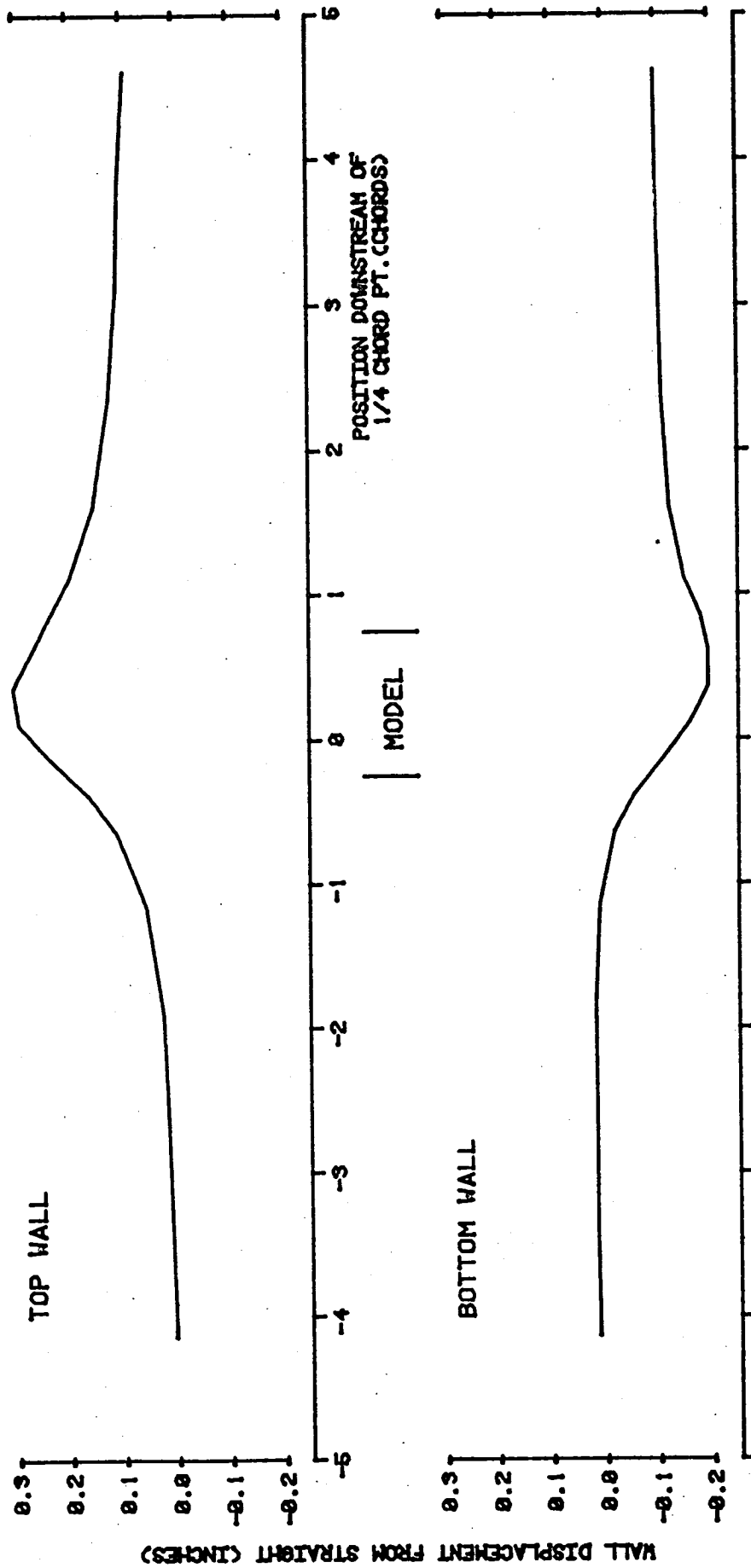


FIGURE 3.2

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
176 2.8 0.891

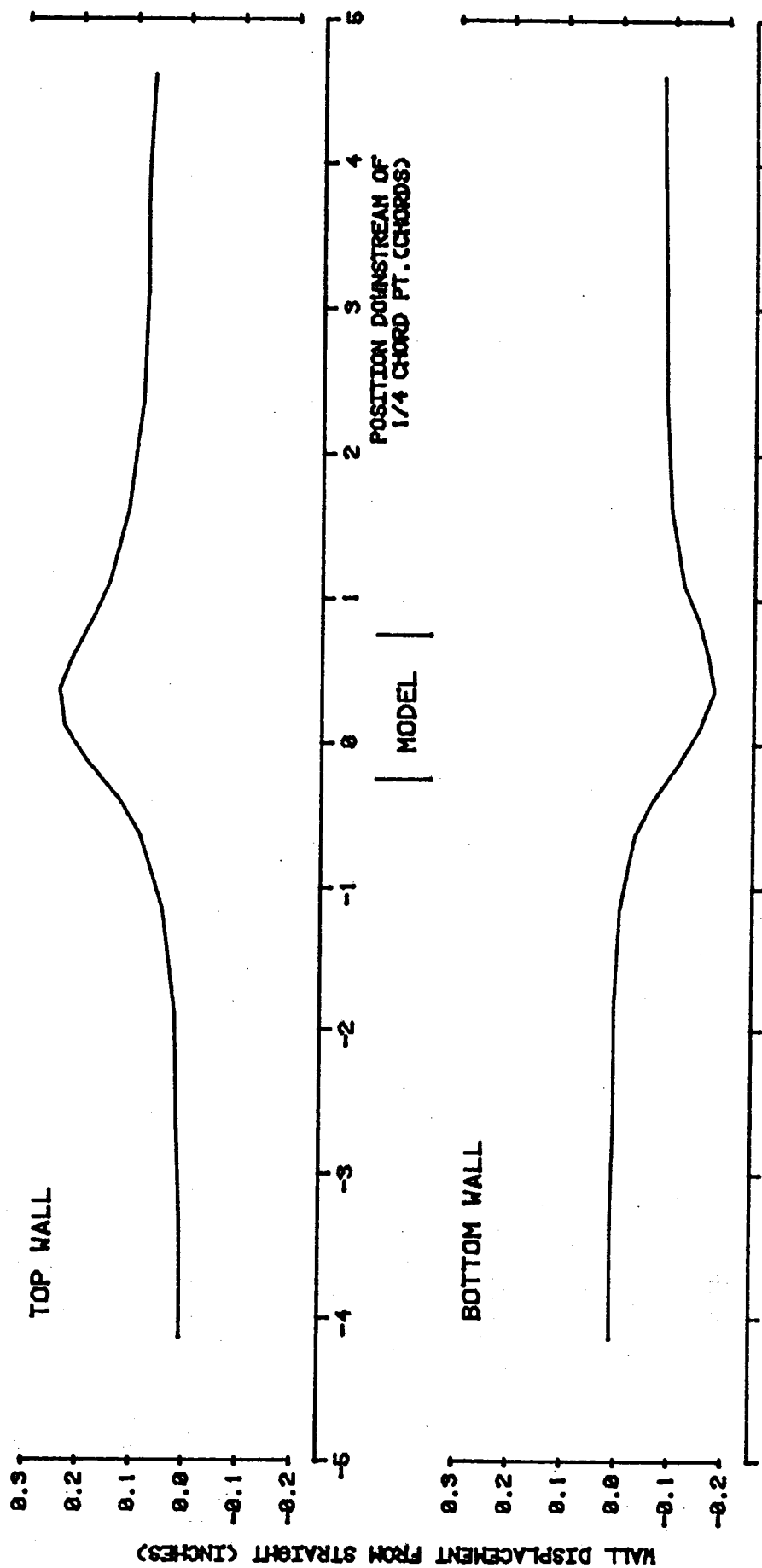


FIGURE 3.3

TSWT WALL DELTA\* CONTOURS

RUN NO 168 ALPHA 0.0 MACH NO 0.868

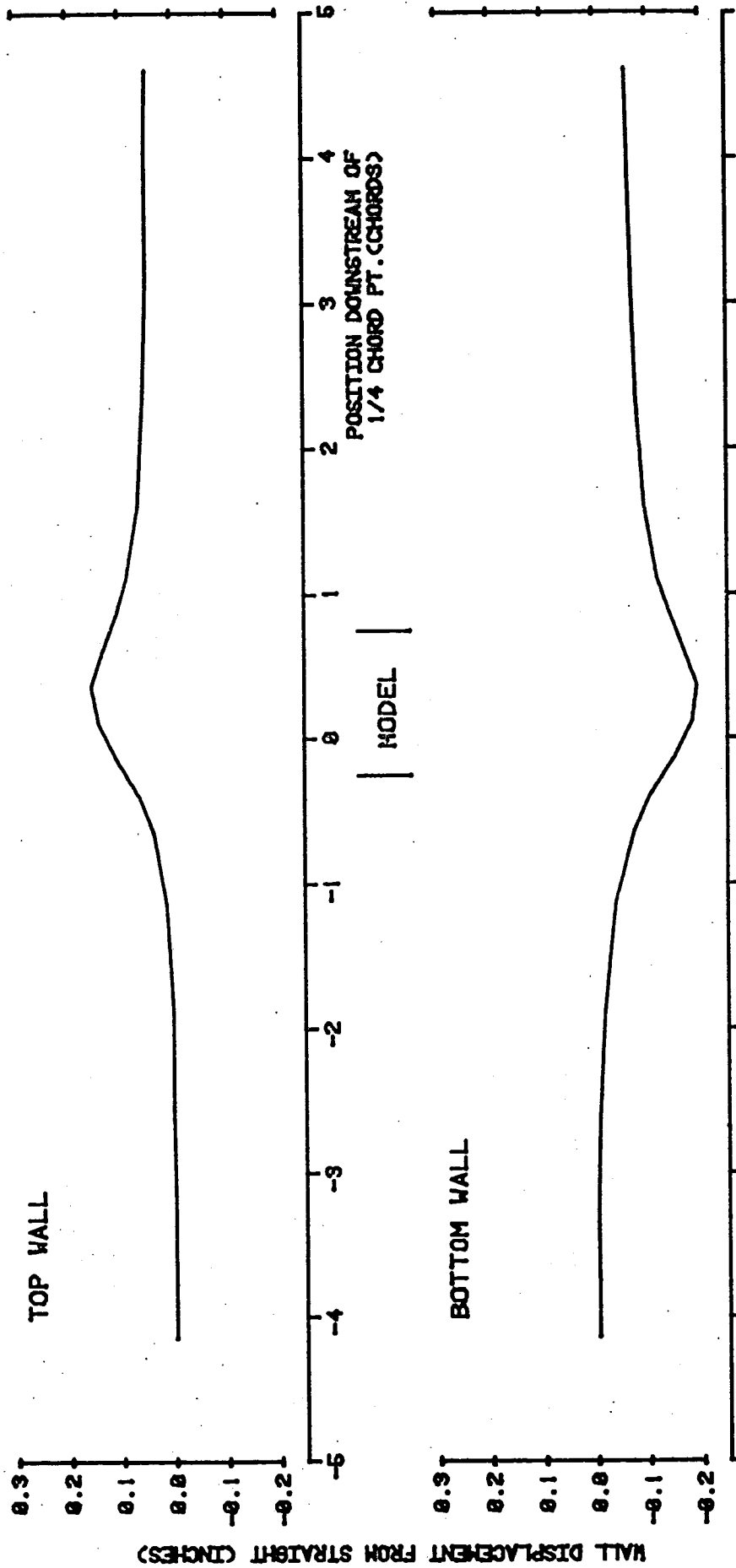


FIGURE 3.4

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
168 4.5 0.846

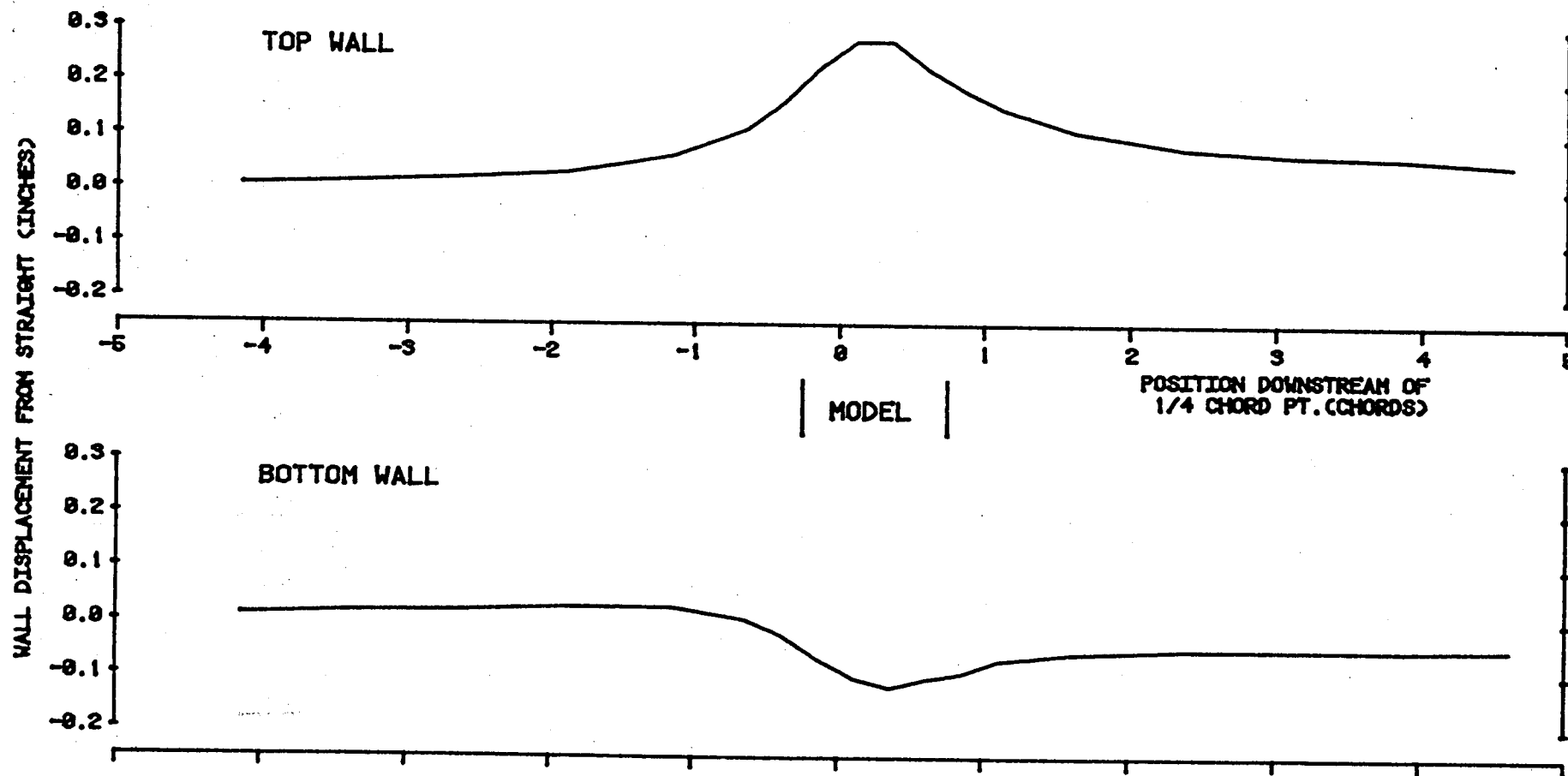


FIGURE 3.5

TSWT WALL DELTA# CONTOURS

RUN NO ALPHA MACH NO  
170 4.5 0.848

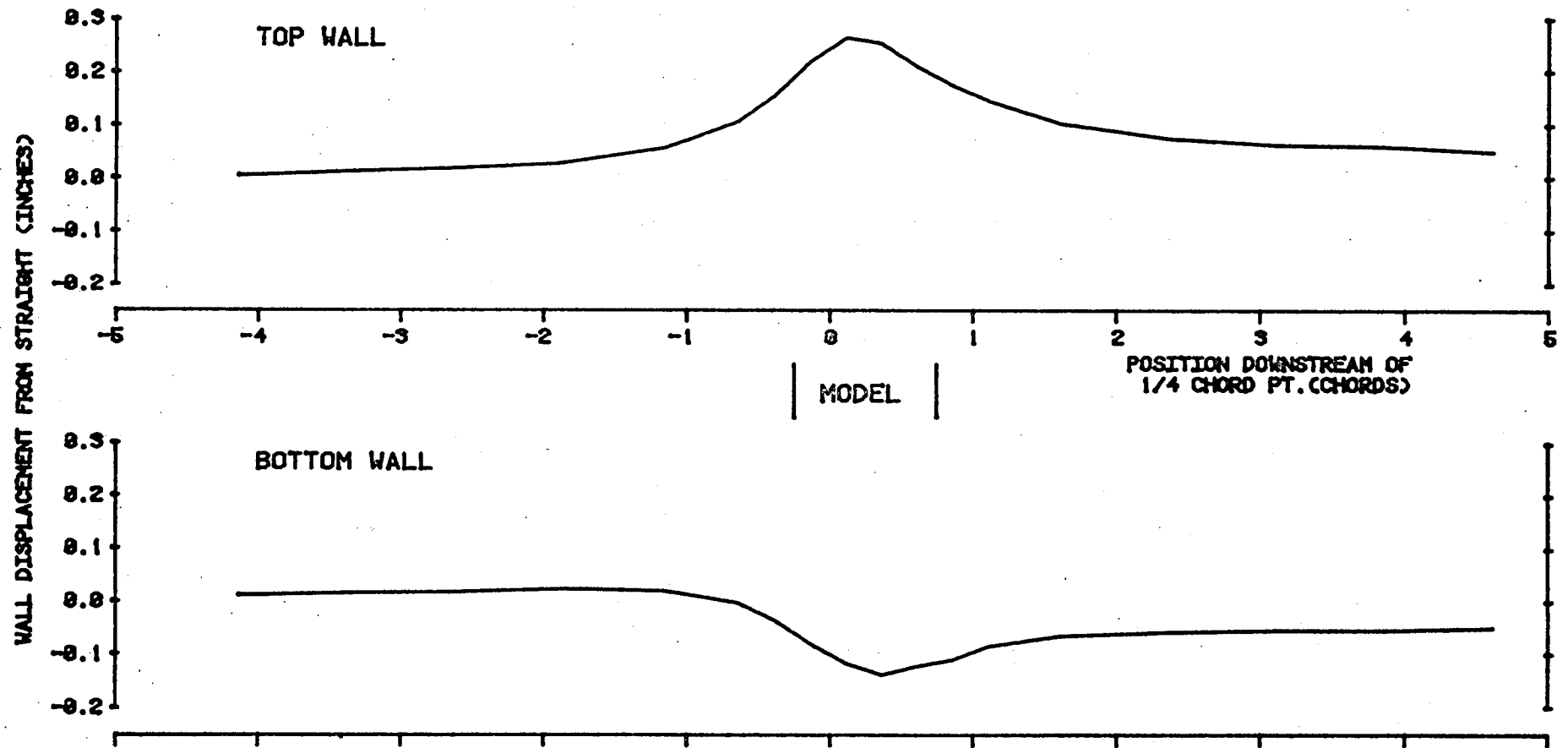


FIGURE 3.6

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
172 2.0 0.848

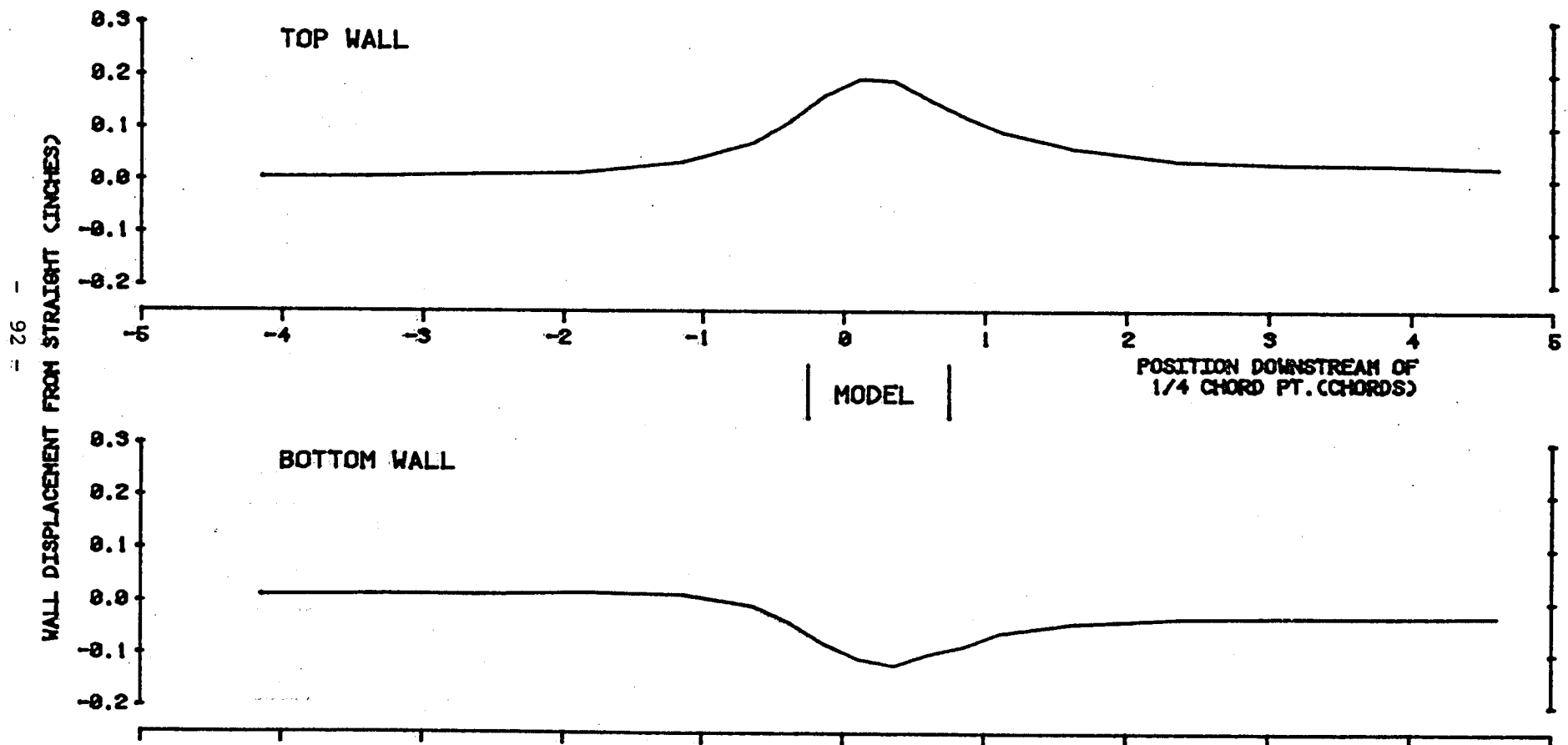


FIGURE 3.7

TSWT WALL DELTA\* CONTOURS

RUN NO 162 ALPHA 2.0 MACH NO 0.838

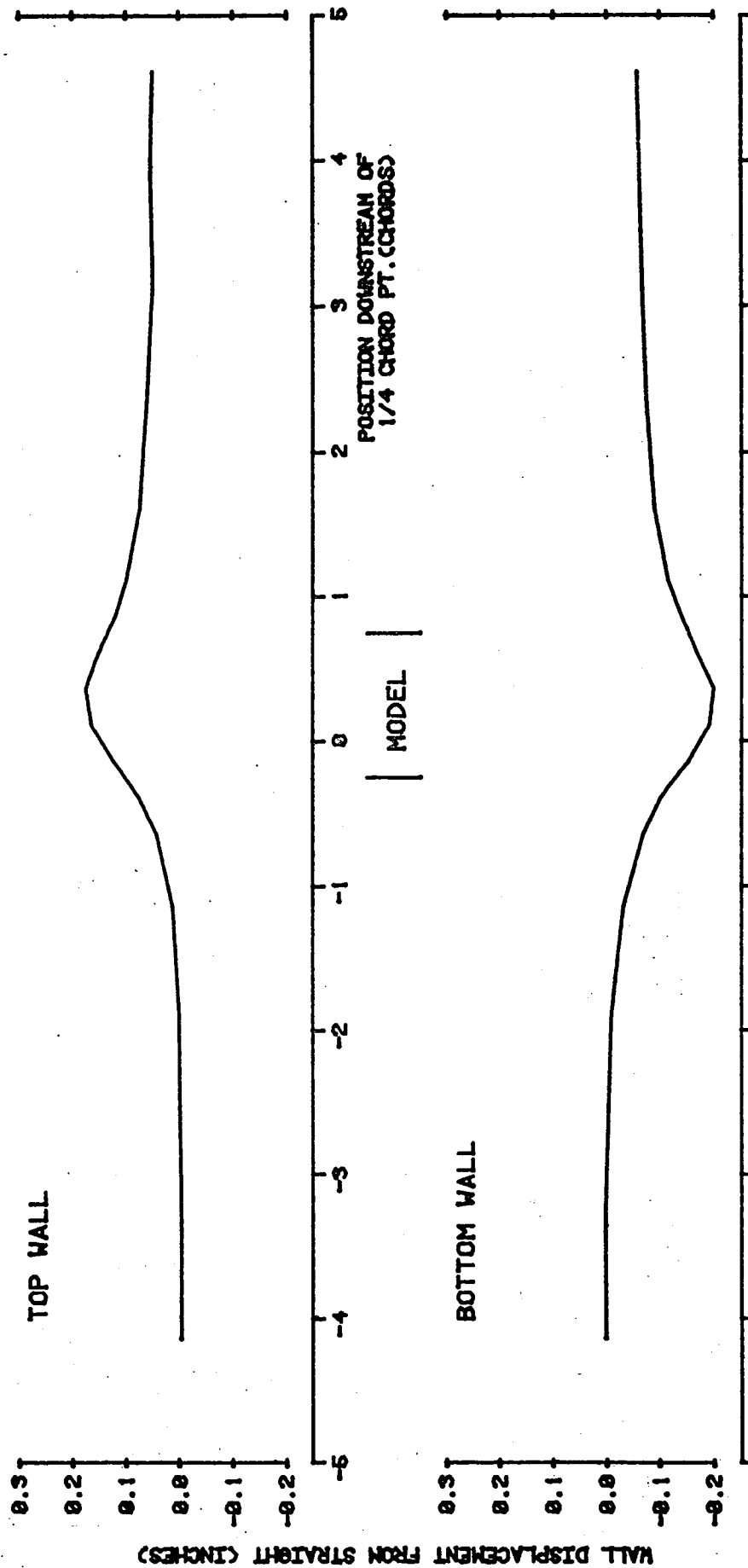


FIGURE 3.8

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
100 2.0 0.840

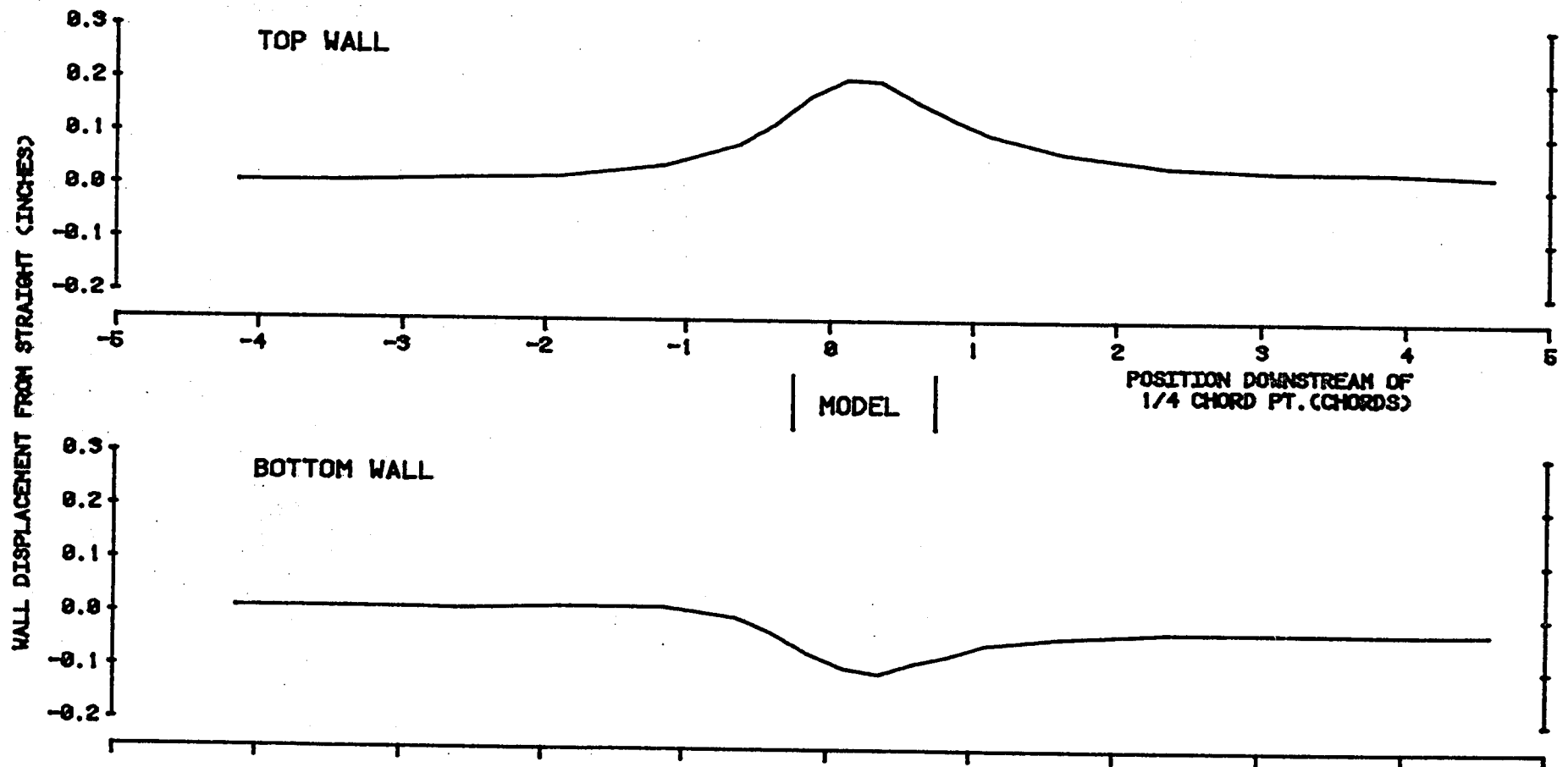


FIGURE 3.9

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
133 0.8 0.854

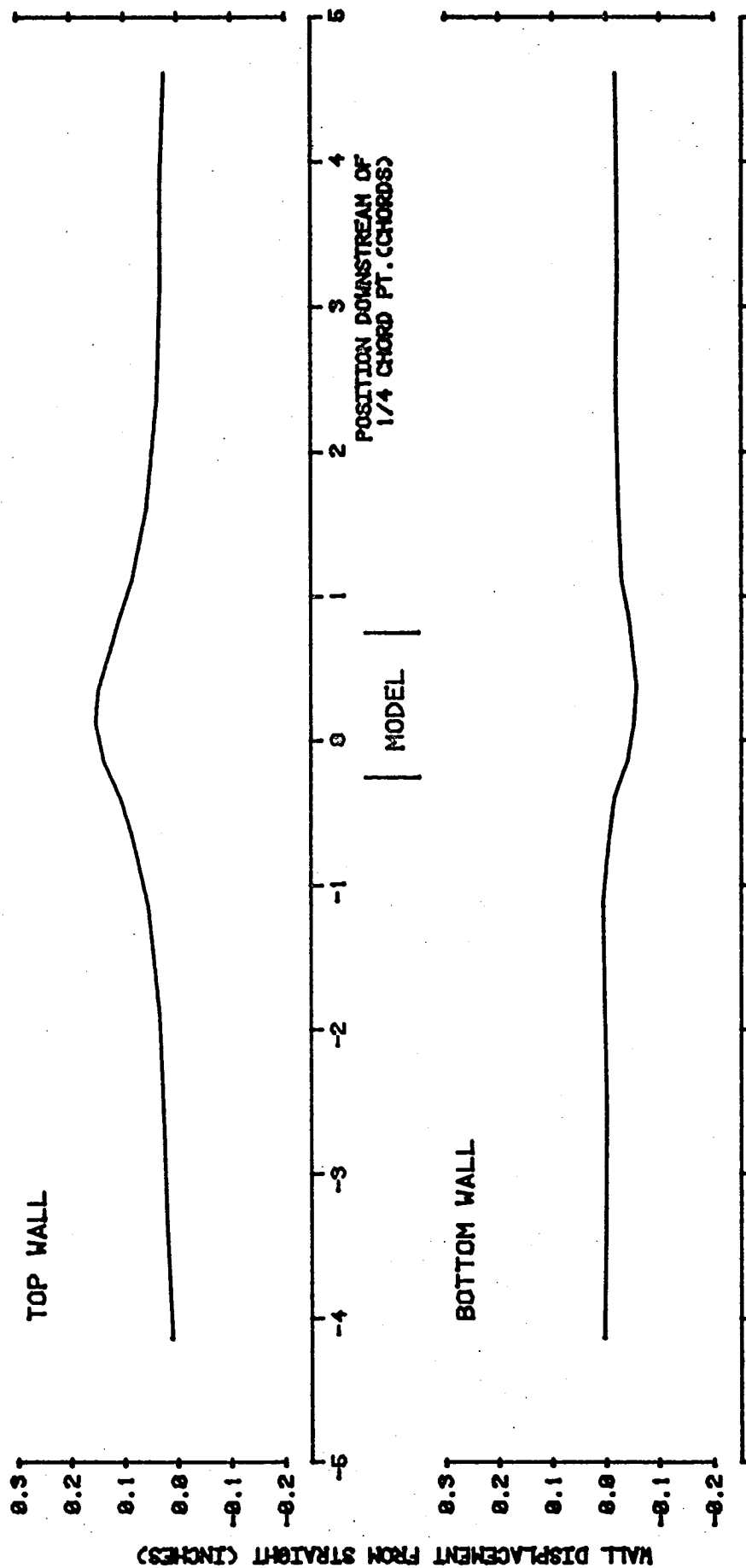


FIGURE 3.10

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
96 2.0 0.806

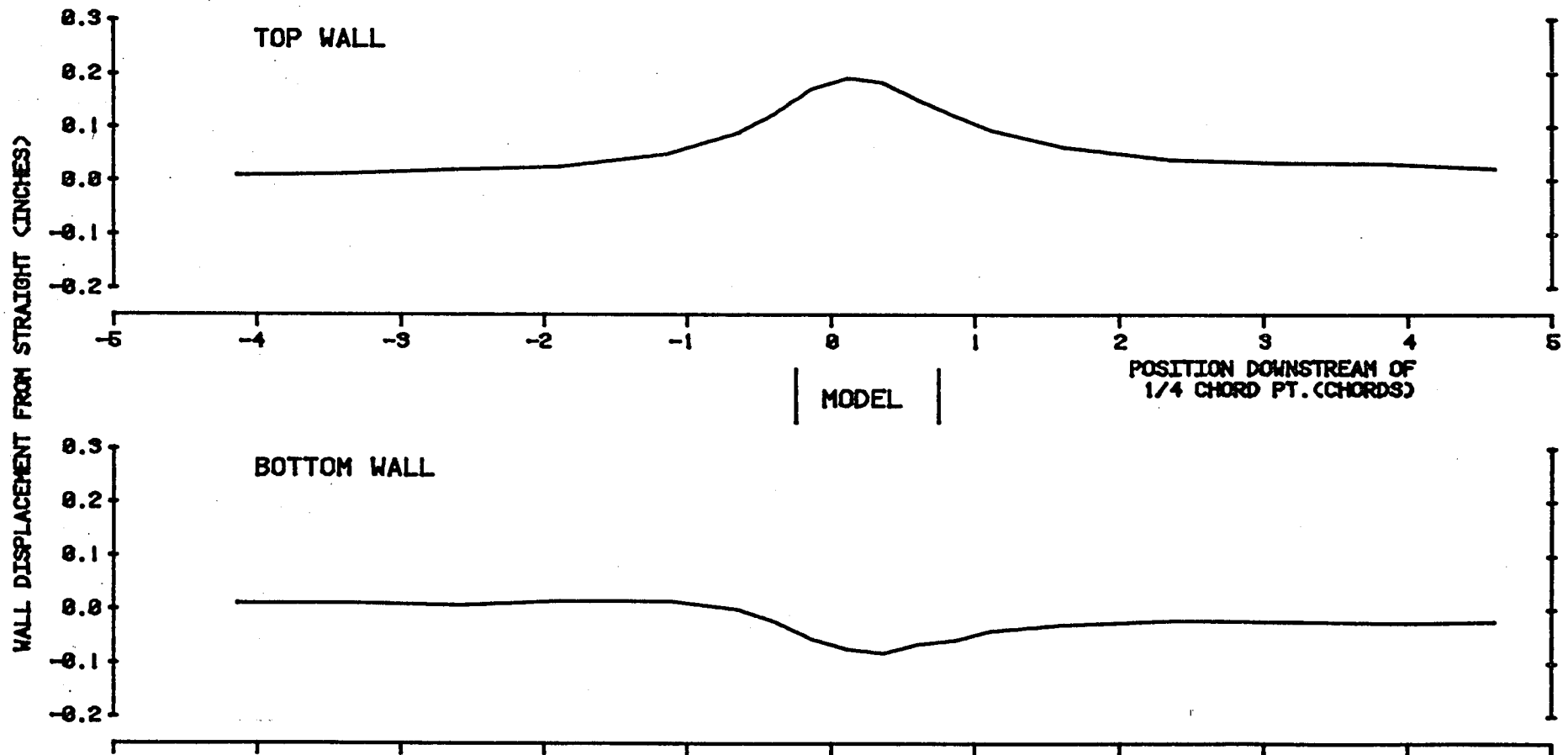


FIGURE 3.12

TSWT WALL DELTA\* CONTOURS

RUN NO 105 ALPHA 0.0 MACH NO 0.753

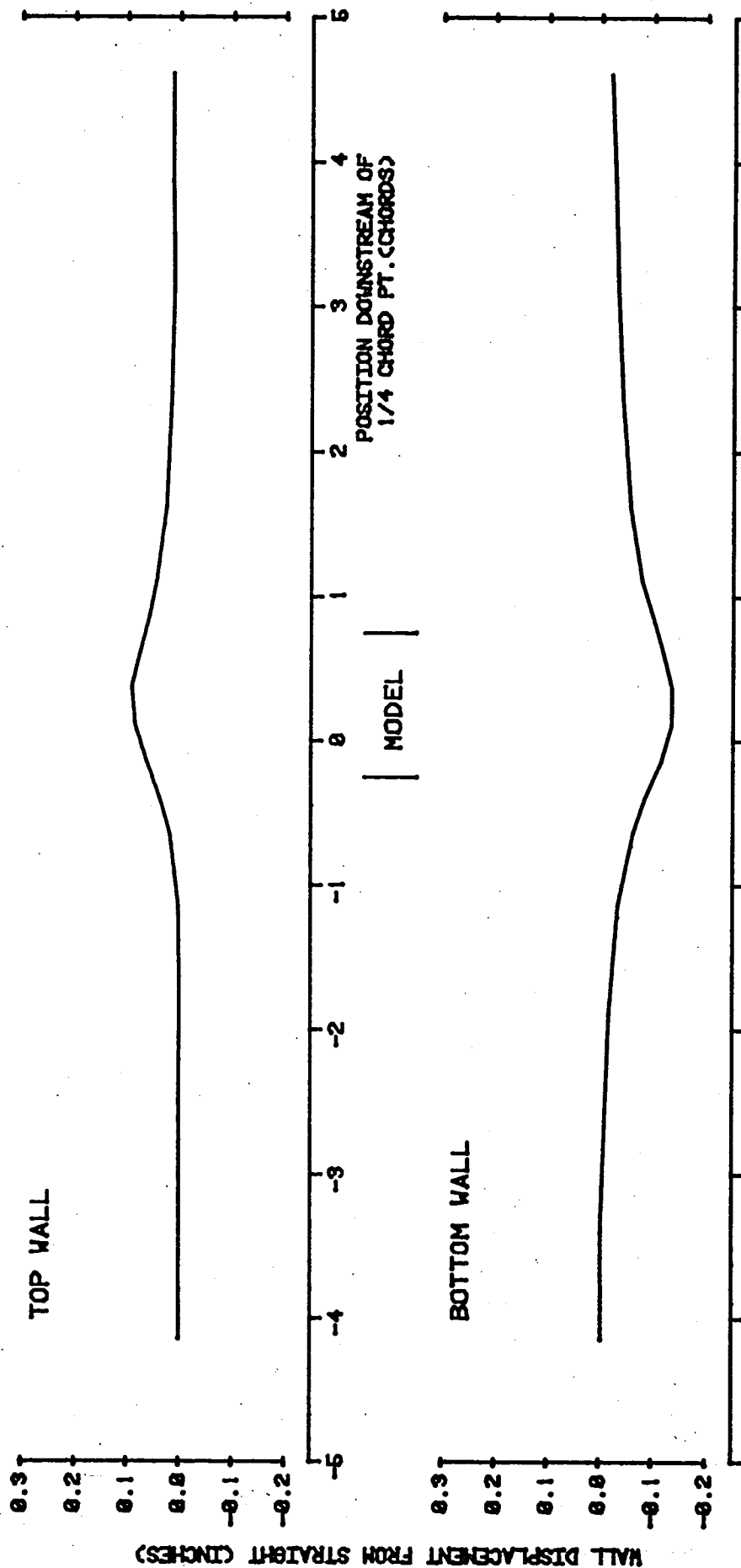


FIGURE 3.17

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
93 2.0 0.712

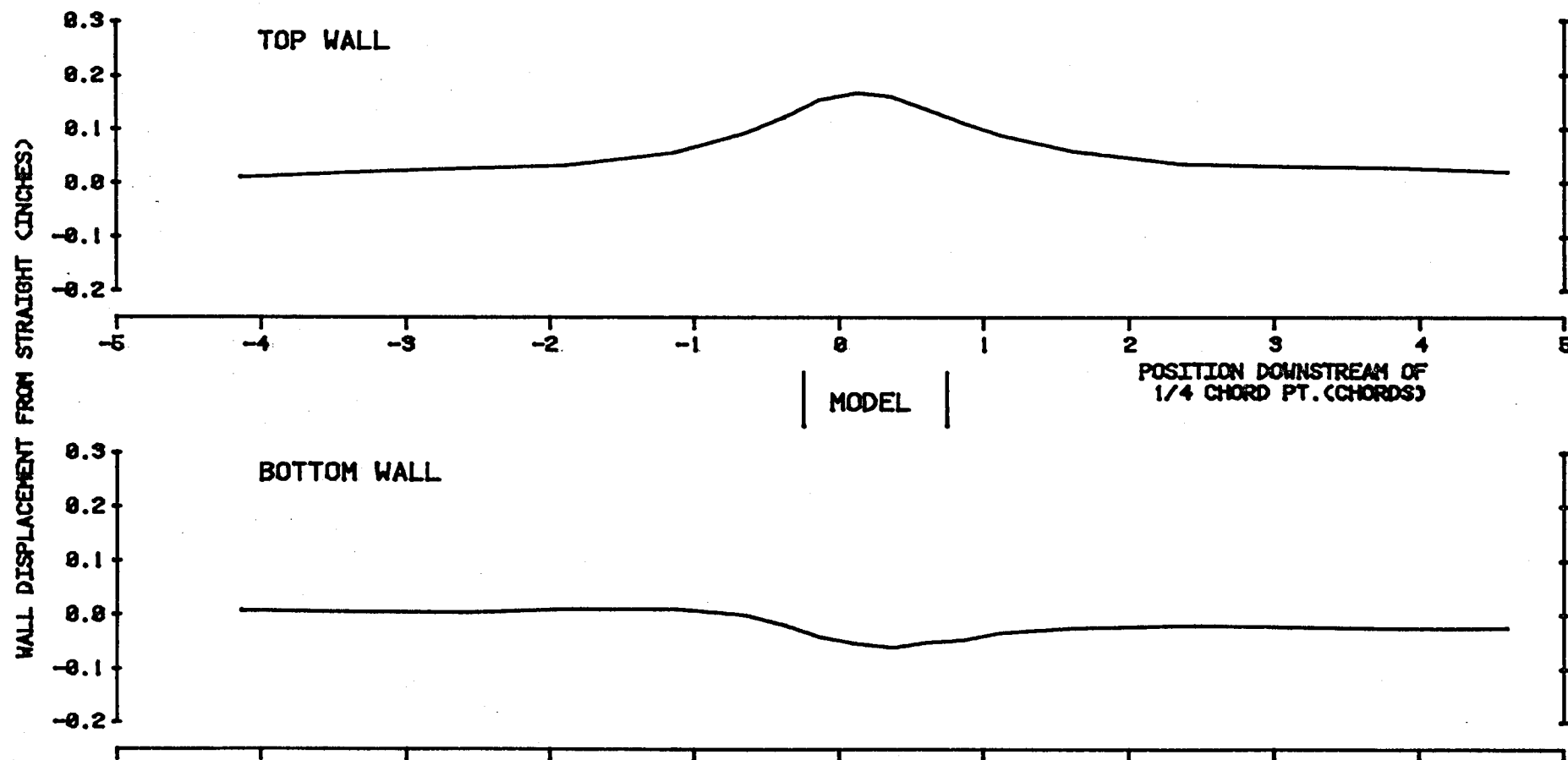


FIGURE 3.18

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
122 0.0 0.698

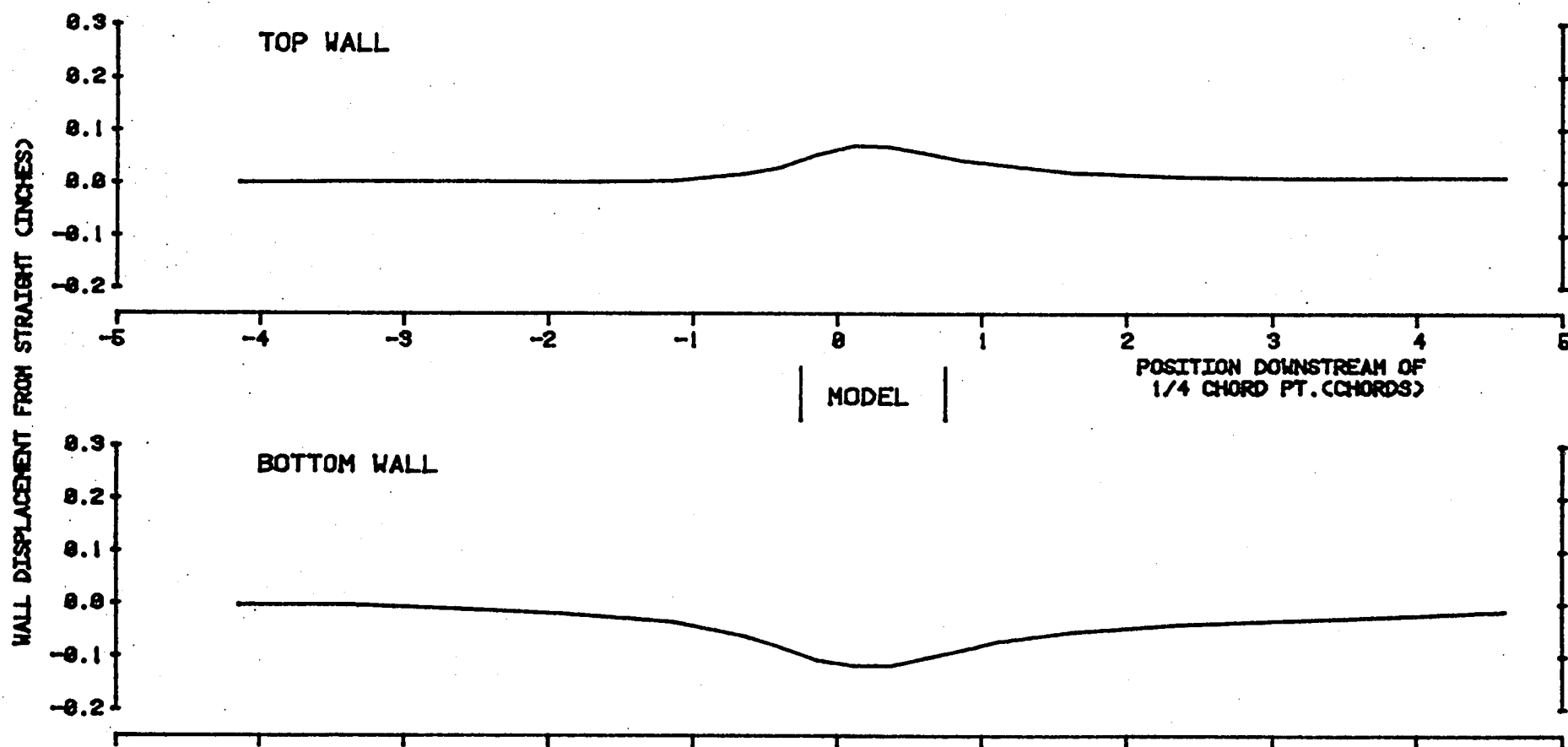


FIGURE 3.19

TSWT WALL DELTA\* CONTOURS

RUN NO 115 ALPHA 6.0 MACH NO 0.506

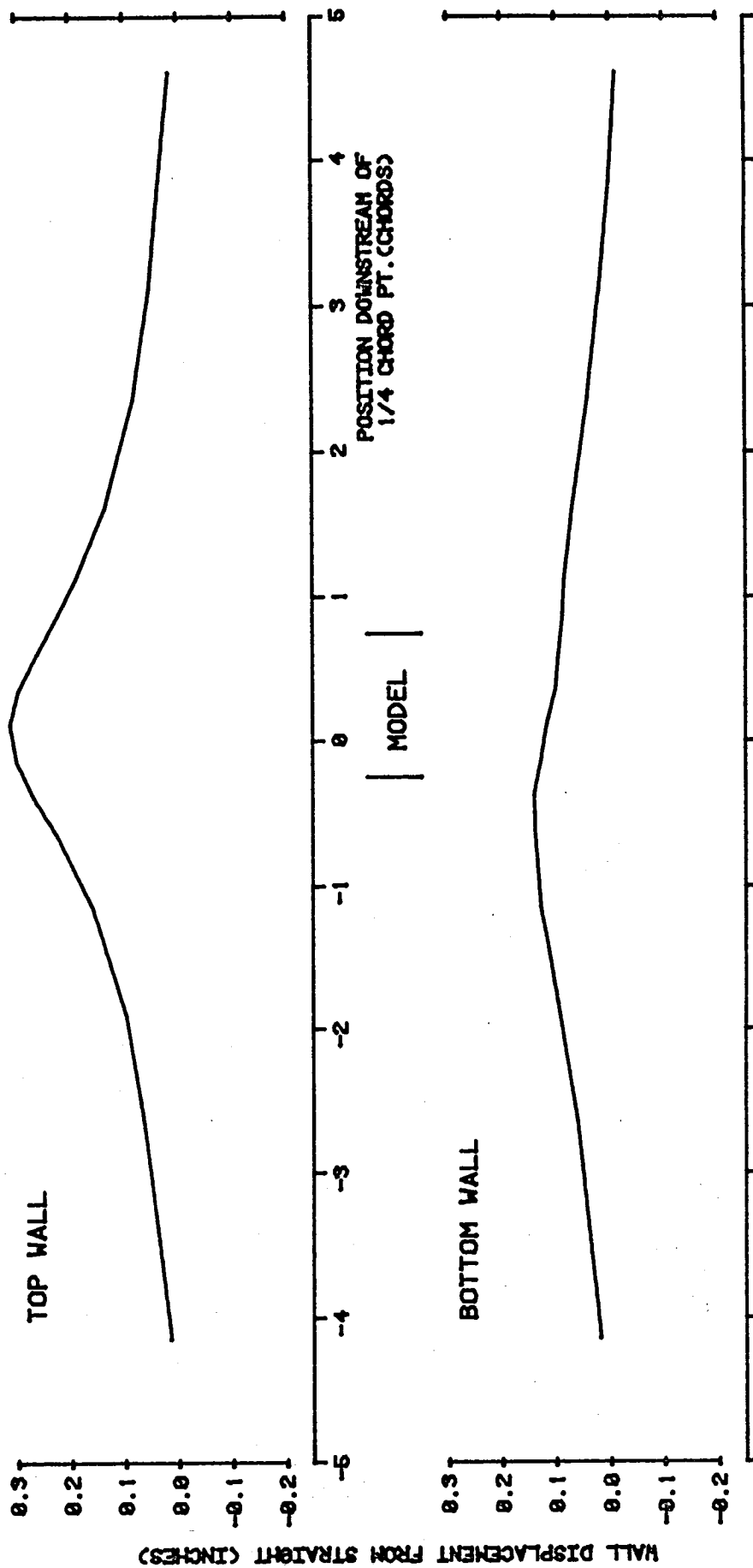


FIGURE 3.20

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
112 4.0 0.507

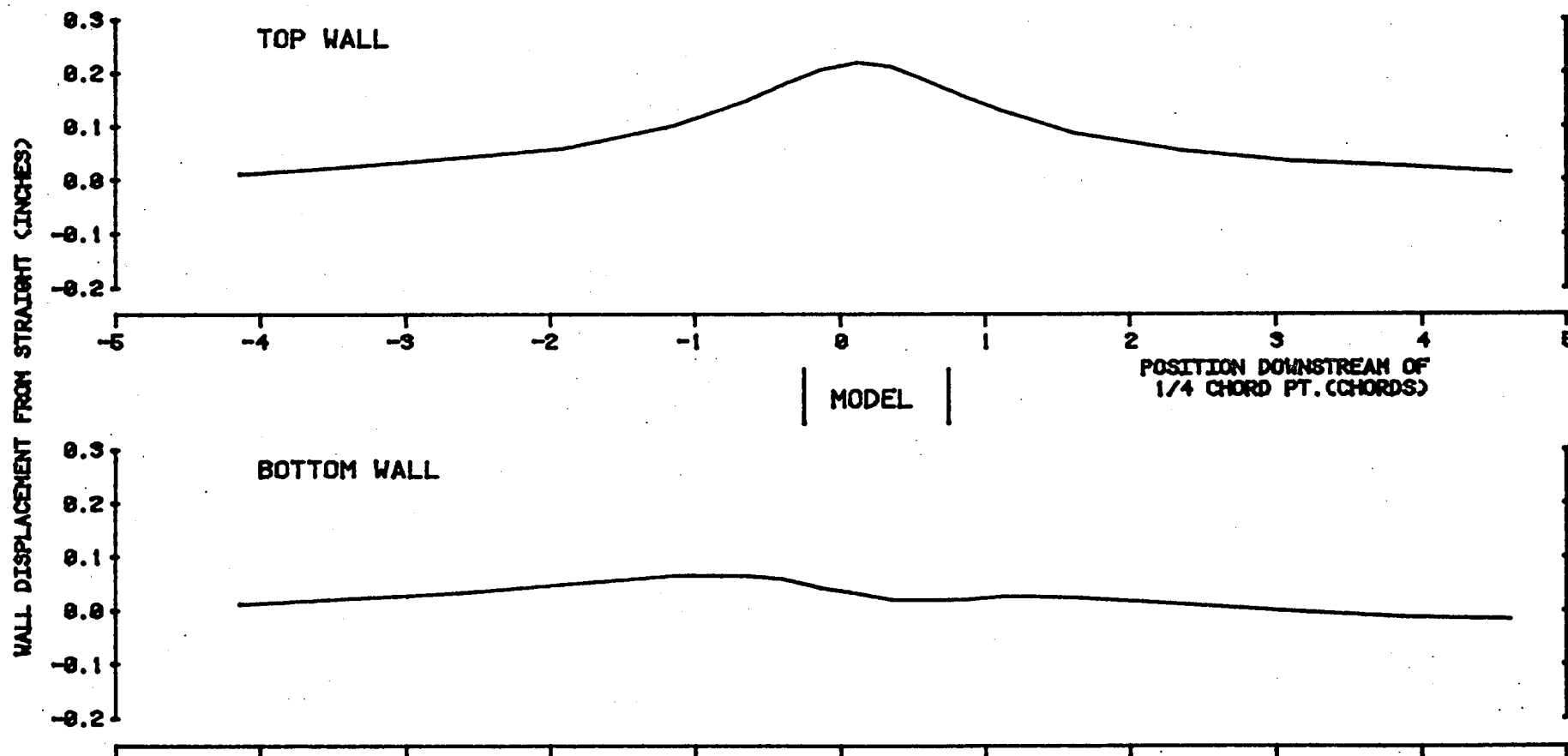


FIGURE 3.21

TSWT WALL DELTA\* CONTOURS

RUN NO 81 ALPHA 2.0 MACH NO 0.508

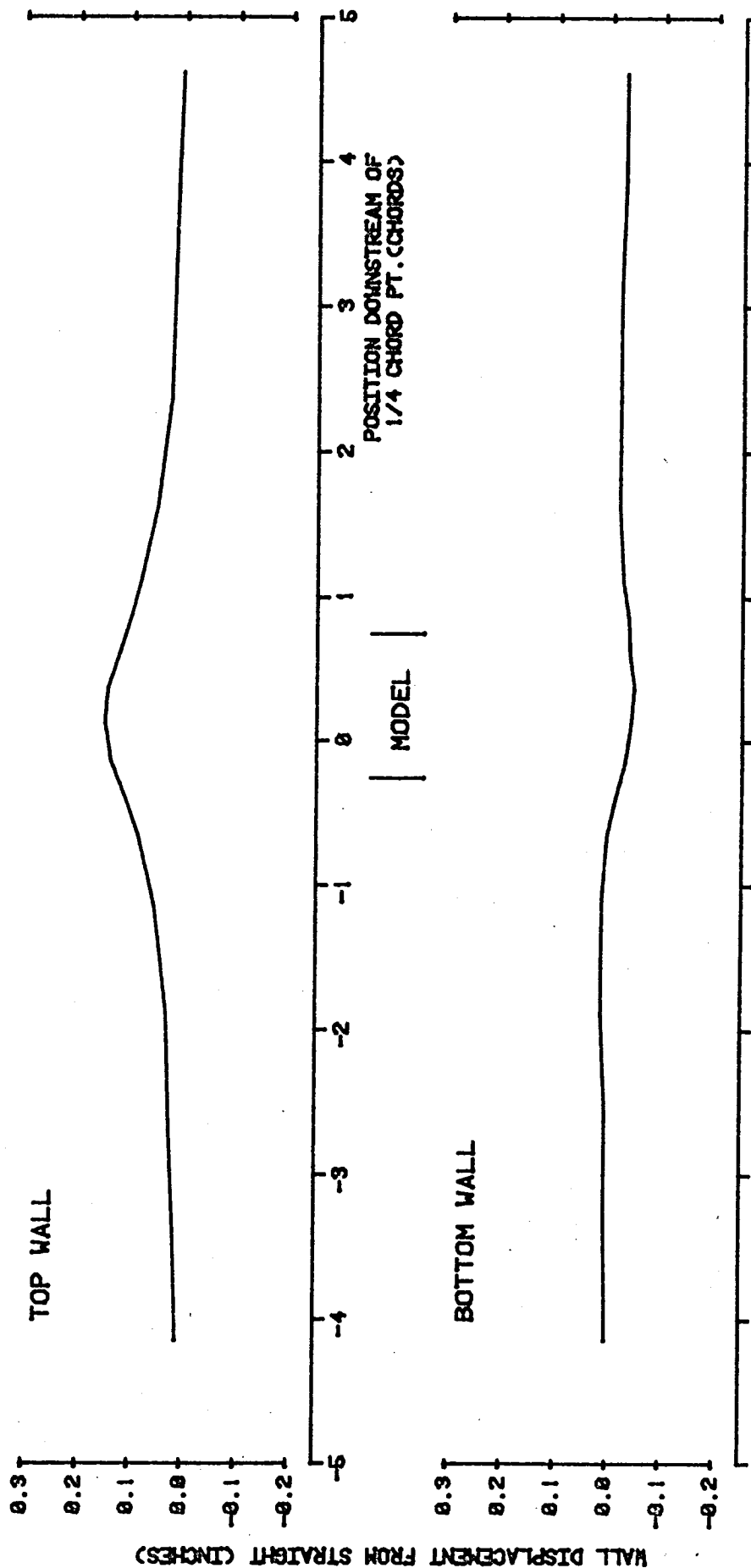


FIGURE 3.22

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
109 2.0 0.504

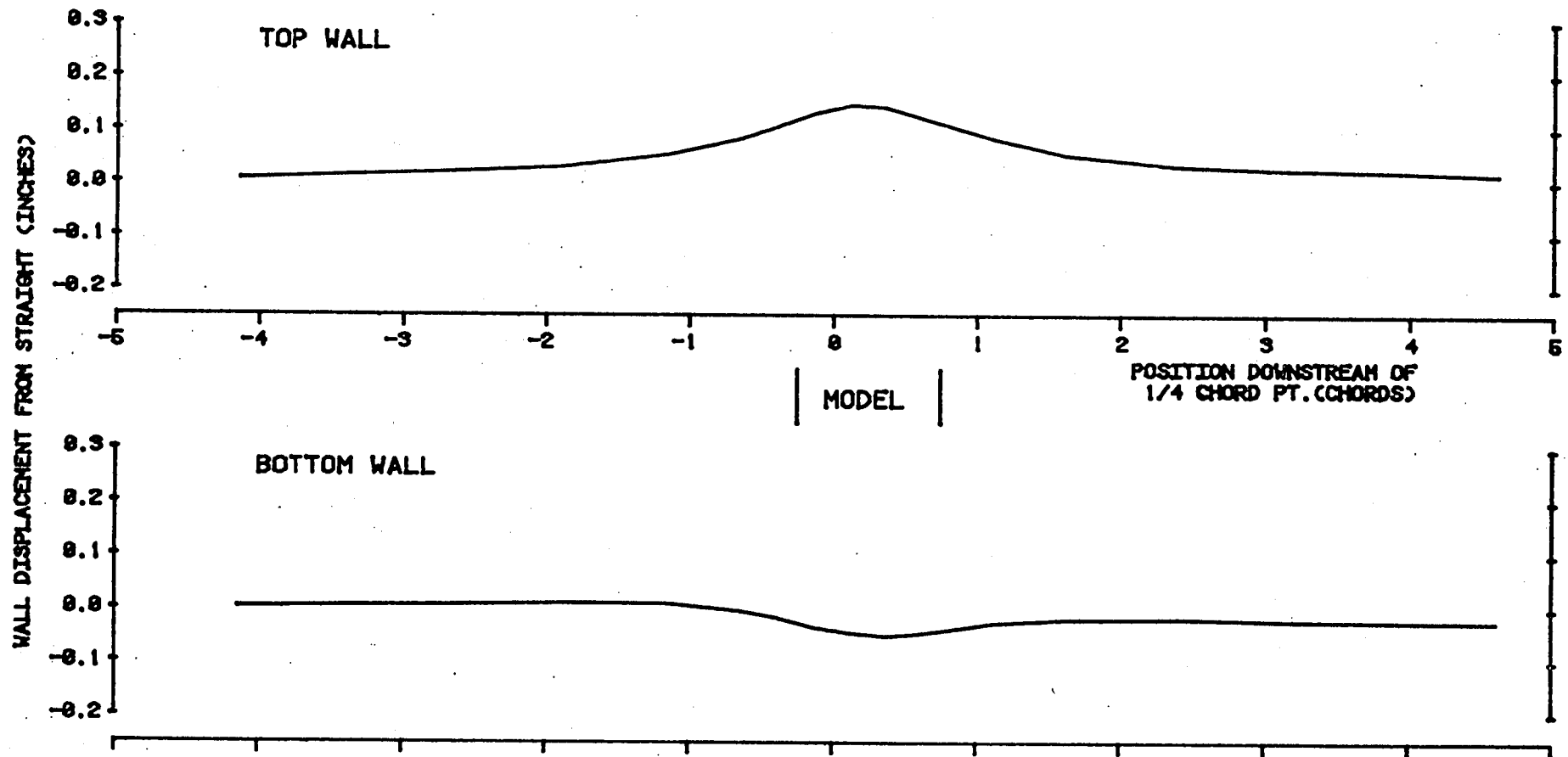


FIGURE 3.23

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
105 0.0 0.506

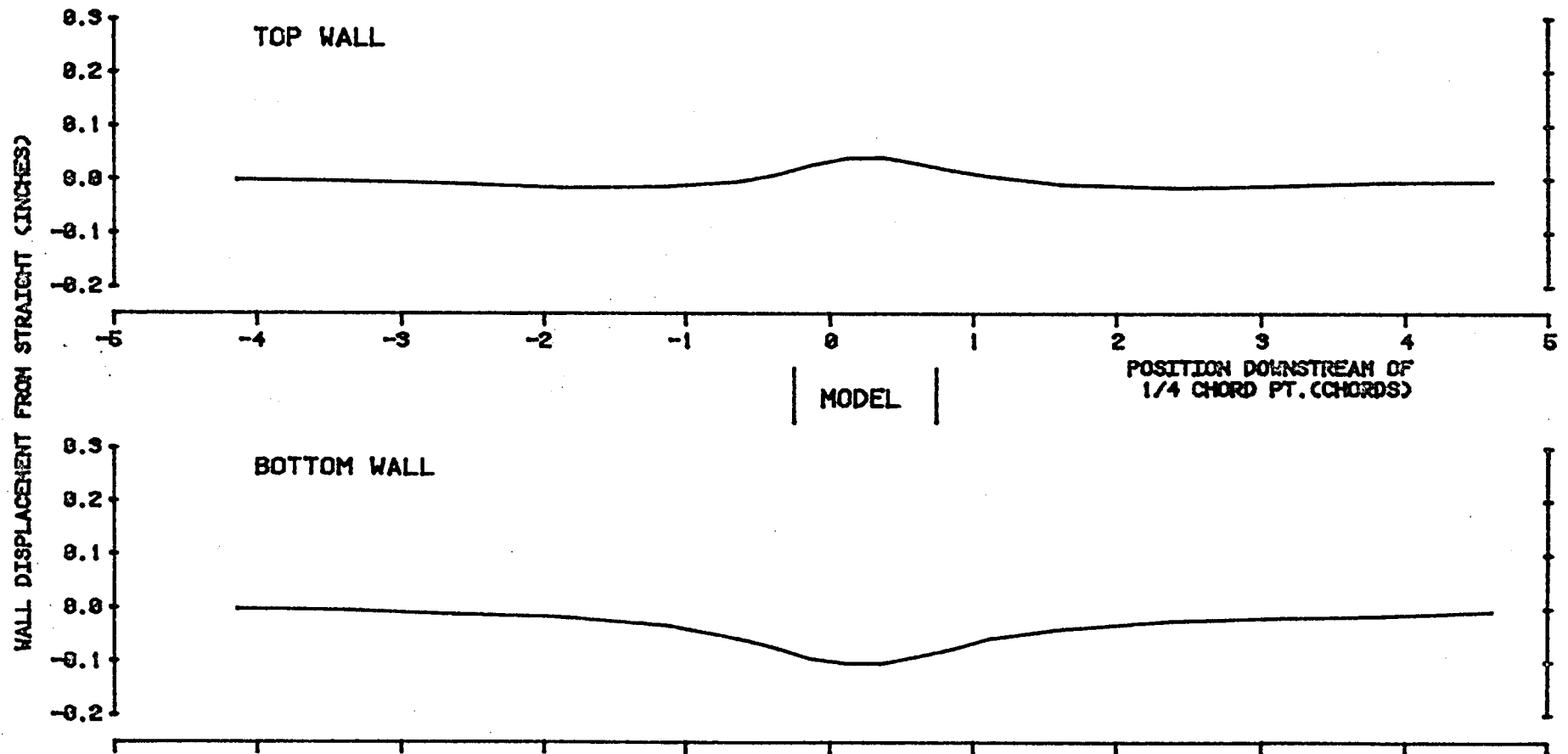
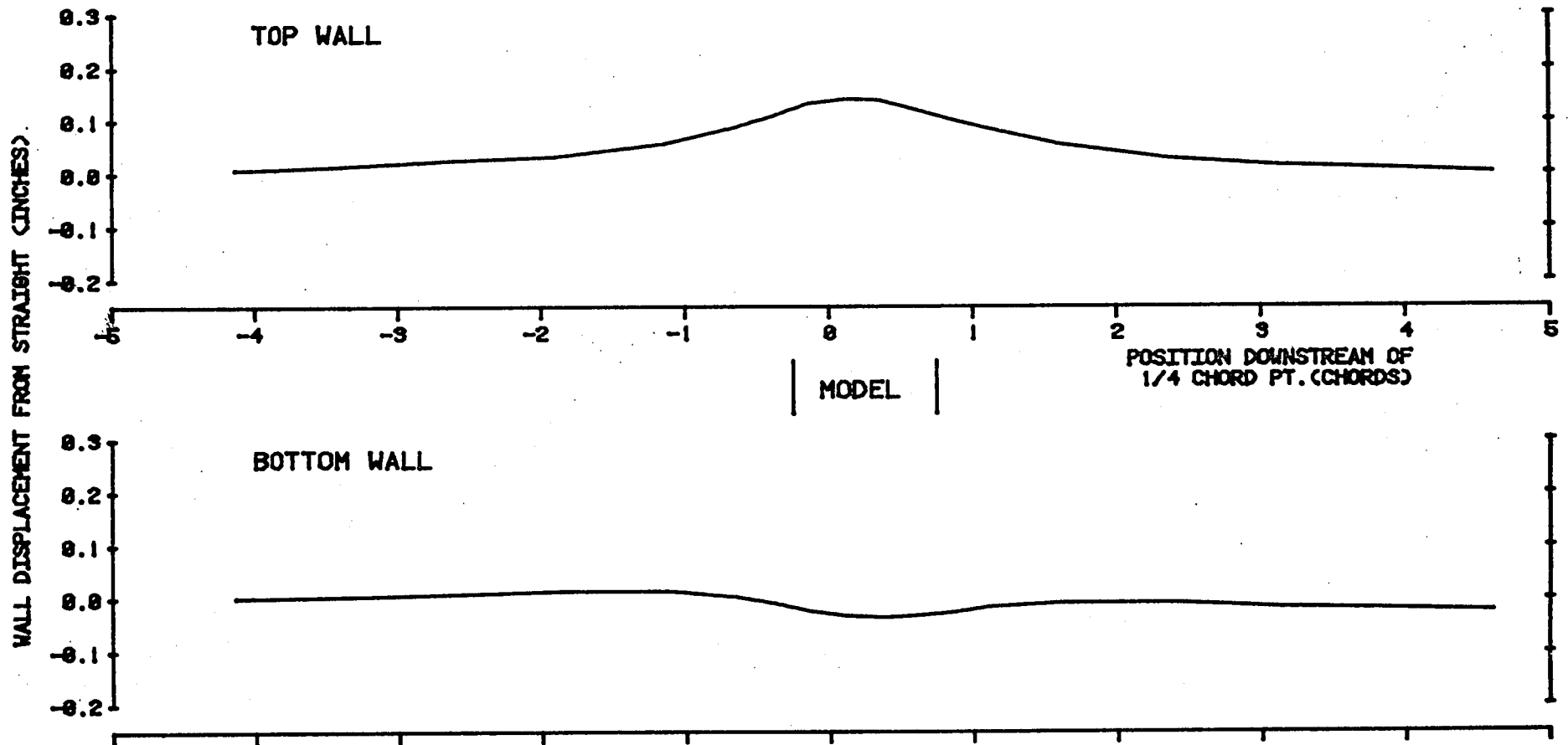


FIGURE 3.24

TSWT WALL DELTA\* CONTOURS

RUN NO ALPHA MACH NO  
89 2.0 0.306



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